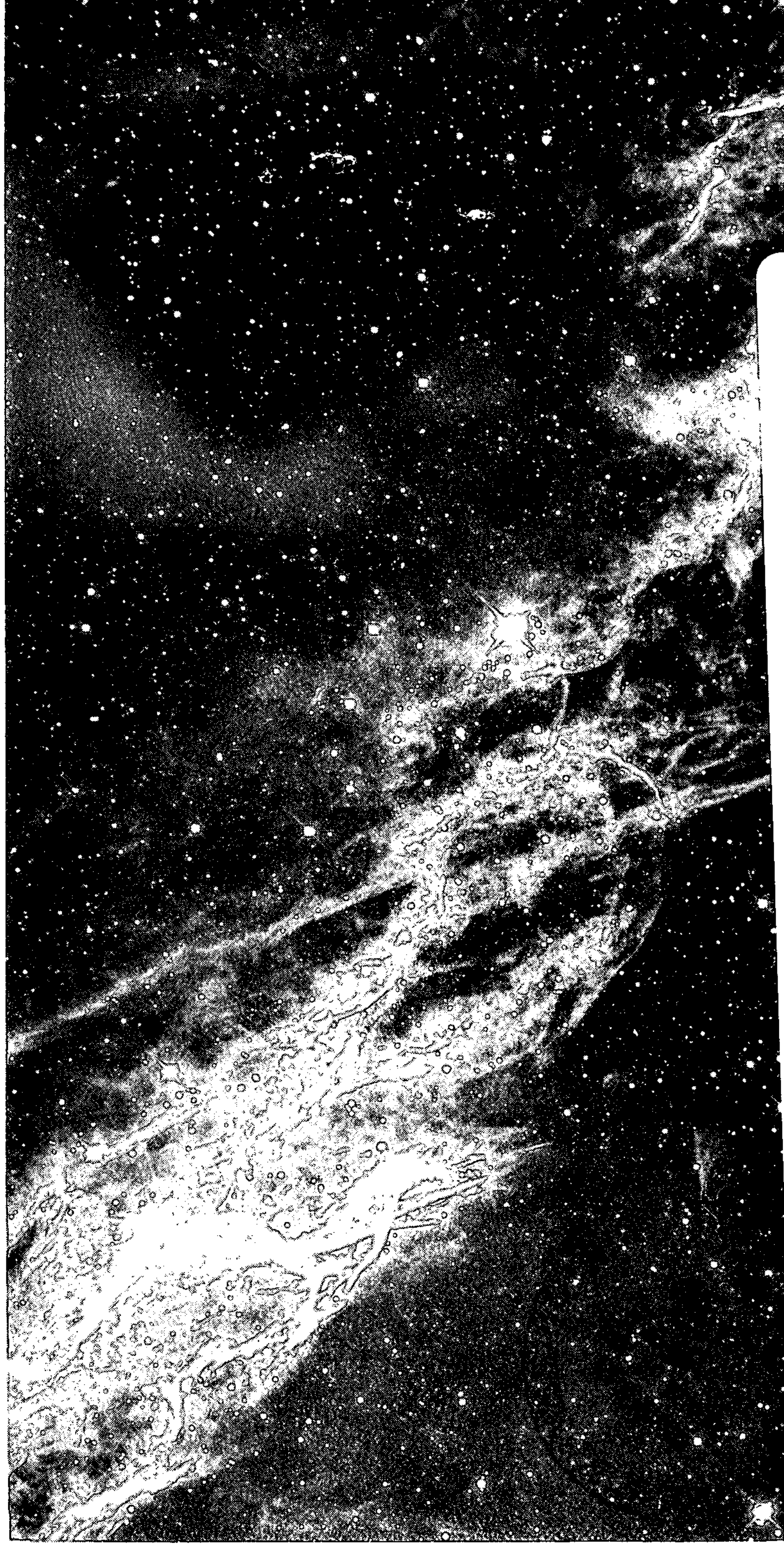
ASTRO
SCIENCES



(NASA-CR-129244) PLANETARY MISSIONS HANDBOOK, FIRST EDITION (IIT Research Inst.) Jul. 1972 98 p CSCL 22A

N73-12887

Unclas
G3/30 16601

NTIS HC $7.00

Technical Memorandum No. T-32

PLANETARY MISSIONS HANDBOOK

(FIRST EDITION)

IIT RESEARCH INSTITUTE
10 West 35 Street
Chicago, Illinois 60616

103p.

Technical Memorandum No. T-32

PLANETARY MISSIONS HANDBOOK

(FIRST EDITION)

by

Astro Sciences
IIT Research Institute
Chicago, Illinois 60616

for

Planetary Programs
Office of Space Sciences
NASA Headquarters
Washington, D. C.

Contract No. NASW-2144

APPROVED BY:

C. A. Stone

C. A. Stone, Director
Physics Research Division

July 1972

TABLE OF CONTENTS

LIST OF FIGURES

LIST OF FIGURES (CONT)

LIST OF TABLES

PLANETARY MISSIONS HANDBOOK

(FIRST EDITION)

1. INTRODUCTION

The purpose of this document is to provide a consistent source of payload performance data for missions to the outer planets. The payload data for both flybys and orbiters is presented graphically as a function of flight time for various launch years, launch vehicles, and orbit sizes. In the past it was necessary to use separate sources for the launch velocity as a function of flight time and for payload as a function of launch velocity. For an orbiter additional calculations were then required to get the net mass in orbit. All the relevant parameters have been combined to produce useful data on a single graph to make advanced mission planning much easier. The limitation of this mode of presentation is that only a reasonable number of payload vs. flight time curves can be drawn. Thus it is necessary to select parameters carefully to reflect frequently used values.

A similar format was used by Horsewood and Mann (1970) who generated optimum solar electric propulsion (SEP) data for flyby and orbiter missions to the planets. However, in order to limit the amount of work, they had to assume a coplanar solar system with circular planetary orbits. While this is a comparatively good assumption for SEP trajectory analysis, performance variations due to launch opportunity are not apparent and a comparison with the ballistic trajectory performance is not easily made.

The next section of the report will discuss the basic assumptions and describe the methods used to calculate the final results. Although the scope of this first edition is limited, covering primarily Jupiter and Saturn missions from 1976 to 1986, the handbook section contains 80 graphs and almost 400 performance curves. Future editions will want to consider a few more launch

opportunities to each of the outer planets and to include more Shuttle based performance options as they become better defined. An appendix, bound separately, contains much of the basic data required in the calculations.

2. METHOD OF ANALYSIS

The mission opportunities included in the first edition of this handbook are given in Table 1. Since the most frequently discussed outer planet missions are those which go to Jupiter or Saturn, emphasis is placed on missions to these two planets in the interval 1976 to 1986. One typical Uranus opportunity, 1985, is also included. Earlier flyby data to Jupiter are included to make up a full cycle of eleven launch opportunities corresponding to one revolution of Jupiter around the sun. Extensive use was made of previously calculated ballistic trajectory data such as that in the Planetary Flight Handbook, NASA SP-35, Part 7 (NASA, 1969). Since it was necessary to calculate new solar electric propulsion (SEP) results, only a limited number of opportunities were studied. However, SEP payloads are less sensitive to launch opportunity.

The procedure for evaluating the performance for ballistic missions involves four specific steps: 1) conversion of the basic trajectory data from tabular into graphical form, 2) interpretation of plotted data in terms of applicable mission constraints, 3) calculation of net useful payload and 4) presentation of the results in the payload versus flight time format. After each of these is described, the SEP calculations will be discussed.

The two sources of ballistic transfer parameters that have been used are SP-35 and computer calculations with the SPARC Program (Roth, et.al., 1968). Only Type I transfers were considered. Both of these sources provide the necessary data; namely, the hyperbolic excess speed at earth departure (VHL), the declination

TABLE 1

MISSION SET FOR FIRST EDITION

OF PLANETARY MISSIONS HANDBOOK

PLANET	MISSION MODES	LAUNCH YEARS	FLIGHT MODES
Jupiter	Flyby	1974-1986	Ballistic
	Orbiter	1976-1986	Ballistic
		1980,83,85	SEP
Saturn	Flyby and Orbiter	1976-1986	Ballistic
		1980,82,85	SEP
Uranus	Flyby and Orbiter	1985	Ballistic and SEP

of the hyperbolic departure asymptote (DLA), the hyperbolic excess speed at planet arrival (VHP) and the declination of the hyperbolic approach asymptote (DEC). VHL is the basic measure of launch vehicle payload performance, the DLA is needed to assess payload degradation due to non-optimum launch conditions. For orbiter missions the VHP is needed to determine what velocity increment is required for orbit capture. While DEC is not used to generate payload results, it is included for completeness and is an important parameter in designing equatorial orbital missions. VHL, DLA and DEC are each plotted against launch date, for fixed arrival date which is essentially the same as flight time, but VHP is such a weak function of launch date that it can be plotted directly as a function of arrival date. For example, the VHL and DLA plot for the 1980 Saturn opportunity are given in Figures 1 and 2. The graphical data for all opportunities are given in an appendix which is bound separately.

The second step toward the objective of payload versus flight time curves is a summary interpretation of the graphical data. Two conditions were imposed on the results during this process. First the launch window was chosen as twenty days, allowing a reasonable time for rescheduling the launch because of minor malfunctions or weather. For example, the maximum VHL over a 20 day window for trajectories arriving at Saturn on 2446470 is 12.39 km/sec which is somewhat greater than the minimum VHL which is 11.38 km/sec. The second condition is a maximum DLA of 40°. Under current range safety and operating practices, missions with a DLA of more than 40°, either north or south, are not launched at the Eastern Test Range where planetary missions are usually initiated. For the example discussed above this requires that the launch not occur before 2444581 so that the maximum VHL increases to 12.75 km/sec on 2444601, the day the window closes. This restriction has not been applied to Shuttle launches.

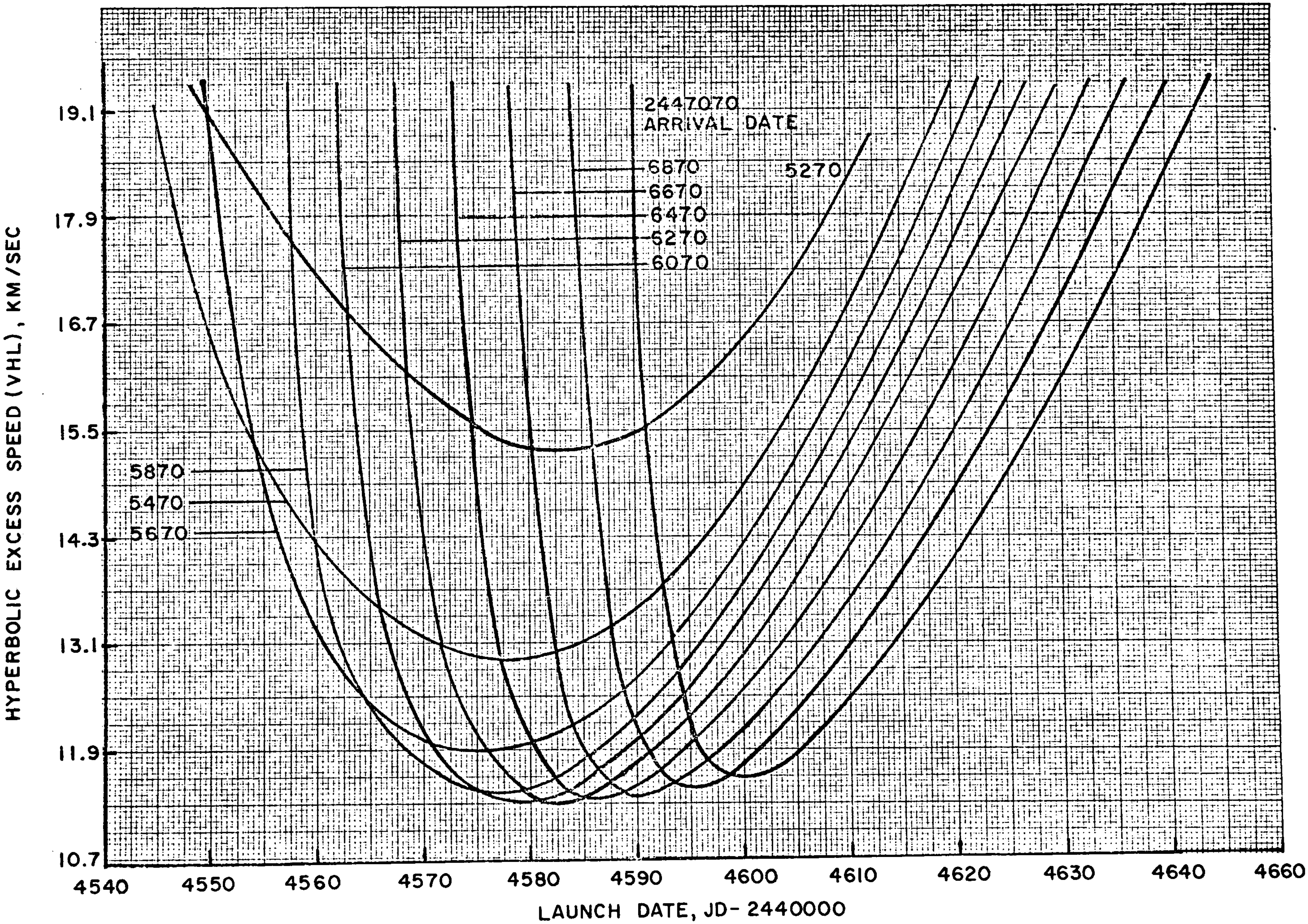


FIGURE I. 1980 SATURN MISSIONS: VHL VERSUS LAUNCH DATE FOR FIXED ARRIVAL DATES

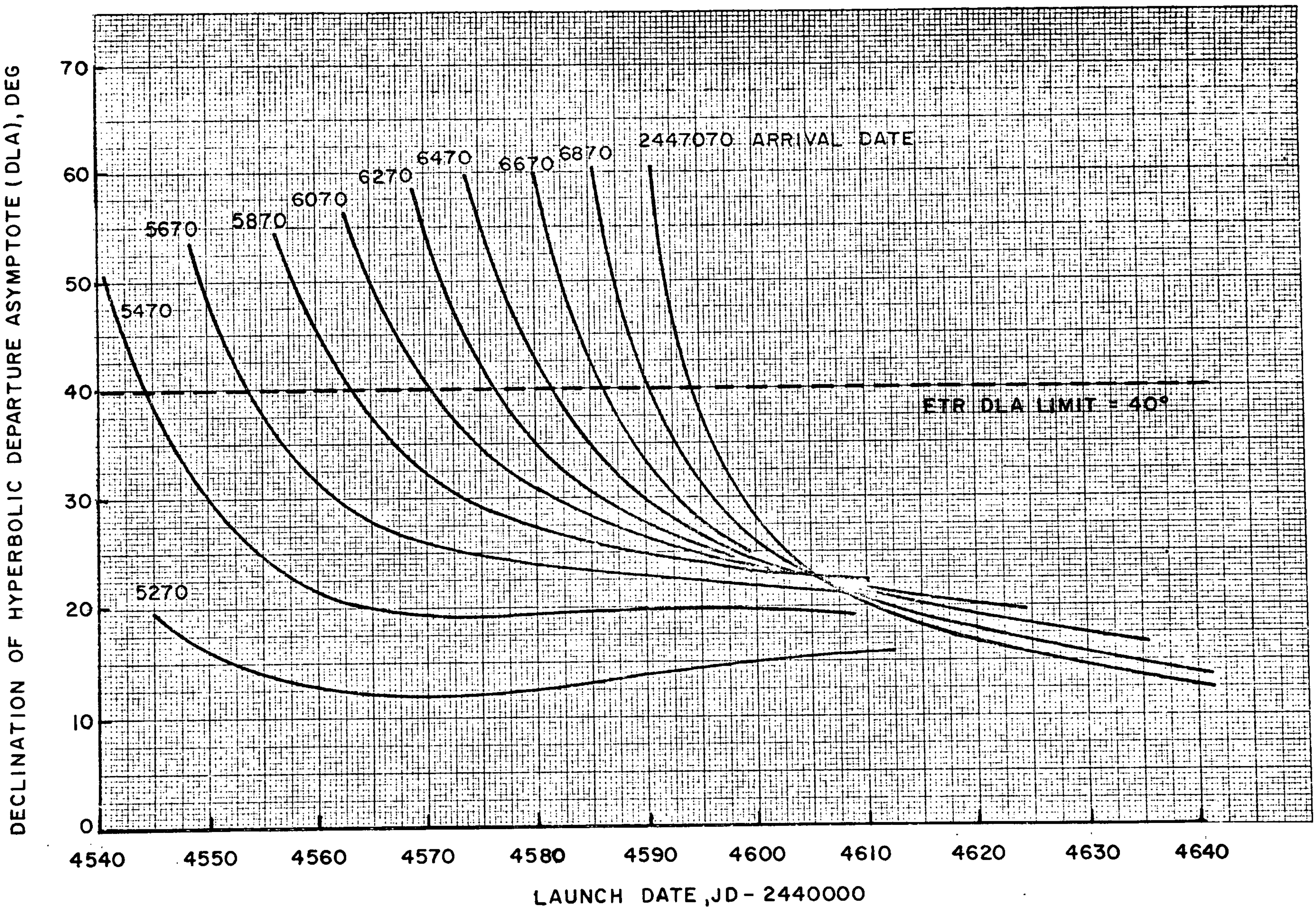


FIGURE 2. 1980 SATURN MISSIONS: DLA VERSUS LAUNCH DATE FOR FIXED ARRIVAL DATES

The next step (and the last one for flyby missions) is to calculate the mass which can be injected to the required VHL by a particular launch vehicle. The launch vehicles which were considered are listed in Table 2. The Titan III E/Centaur vehicle will be used for the Viking missions and will be available in 1974 for outer planet missions. The Shuttle is being developed for use in the 1980's. The performance of the Shuttle/Centaur has been estimated, but a new upper stage may be developed. The basic data of launch vehicle performance (or payload) vs. characteristic velocity*, V_c, was taken from the Launch Vehicle Estimating Factors document (NASA 1972). This graphical data was converted to analytical form by fitting it to an equation of the form

$$\text{Payload} = B_1 \exp(-V_c/B_2) - B_3$$

The injected payload for flyby missions is easily obtained since all quantities on the right hand side are known for a particular flight time and launch vehicle.

For orbiter missions the velocity increment (ΔV) required to capture into the desired orbit can be easily calculated using the planet gravitational mass, the periapse radius, the orbit period and the VHP. Table 2 also contains a list of candidate planetary orbits. The periapse for Jupiter was set at 4.0 planet radii because of the radiation damage problem. Saturn's rings dictate the selection of 3.0 planet radii as the periapse distance, while at Uranus a low periapse appears safe. Several orbit periods were selected for each planet to span a range of apoapse distance and of ΔV requirement. Through 1980 an earth-storable retro propulsion system with specific impulse (I_{sp}) of 285 sec

---

* $V_c^2 = V_e^2 + VHL^2$ where V_e is the earth escape velocity from a 100 nm circular orbit. A correction is applied for DLA > 28.6° when easterly launches are not possible.

TABLE 2

FIXED PARAMETERS FOR FIRST EDITION OF PLANETARY MISSIONS HANDBOOK

LAUNCH VEHICLES:

1976-1986	Titan III E/Centaur Titan III E/Centaur/BII (2300) Titan III E/Centaur/TE 364-4 Titan III E/Centaur/SEP (20 kw)
1980-1986	Shuttle/Centaur Shuttle/Centaur/HE BII Shuttle/Centaur/SEP (20 kw)

LAUNCH CONDITIONS:

20 day window/DLA < 40° for Titan vehicles

RETRO PROPULSION SYSTEMS:

1976-1980	Earth-Storable, Isp = 285
1980-1986	Space-Storable, Isp = 375

ORBITS:

PLANET	PERIAPSE	PERIODS
Jupiter	4.0 R_J	15,30,60^d
Saturn	3.0 R_S	15,30,60
Uranus	1.2 R_U	5,15,60

was assumed. This corresponds to Viking technology. However, by 1980 the space-storable propulsion technology with an I_{sp} of 375 sec should be ready. Since calculations were made for both types in 1980, it will be possible to compare their relative performance.

In addition to performing the orbit capture manuever, the retro system will also perform midcourse trajectory corrections and orbit trim maneuvers. For this purpose 250 m/sec of additional ΔV capability is provided. Thus when using the payload curves, the midcourse propulsion system should be included in the mass of flyby spacecraft but excluded from orbiters since it has been included in the orbit capture propulsion system.

The net mass in orbit is now easily calculated using the flyby approach mass, the ΔV and the I_{sp} according to the empirical formulation of Chadwick (1968). All of the above payload calculations and the last step, the plotting, can be done with a Hewlett-Packard desk calculator system.

No tabular source of data comparable to the SP-35 ballistic trajectory handbooks exists for solar electric low-thrust trajectories. Therefore calculations were made using the Chebytop computer program (Hahn and Johnson, 1971). The launch years for which the SEP mode was considered are listed in Table 1. All calculations were made with a fixed thruster specific impulse of 3000 sec which is somewhat higher than optimum for outer planet missions, but is the technologically preferred value. Values of the net payload and solar electric power level, both normalized to 1000 kg of mass injected to earth escape, were found for fixed thruster on-times between 200 and 600 days. The data presented in this handbook is for a 400 day propulsion on-time, but the results are not very sensitive to this parameter. Cutting the thrust off at 400 days means that there will generally be sufficient power for a 3.0 kw thruster with a 4.1 throttling capability

after providing 400 watts of housekeeping power for the SEP subsystems.

So far the data are independent of the launch vehicle. They can be scaled to a particular launch vehicle by using its injected mass capability at each of the several VHL values for which calculations were performed. Figure 3 shows the results of the scaling for the 1980 Saturn launch opportunity with net approach mass plotted as a function of the SEP power level. A SEP power level of 20 kw was selected as representative of current studies for outer planet missions. The net payload versus flight time curve can now be drawn by interpolating the payload at 20 kw.

No provision has yet been made for a solar electric stage whose weight would reduce the net payload by an estimated 300 kg. For flyby missions the stage weight should be added to the spacecraft to get the net payload required. For orbiters it was assumed that the SEP system including 300 kg of stage weight would be jettisoned before going into orbit. Thus the orbiter net payload would be somewhat higher if the stage weighed less. The orbit capture calculation and the plotting of the SEP payload versus flight time curves were performed with the same desk calculator system as for the ballistic curves.

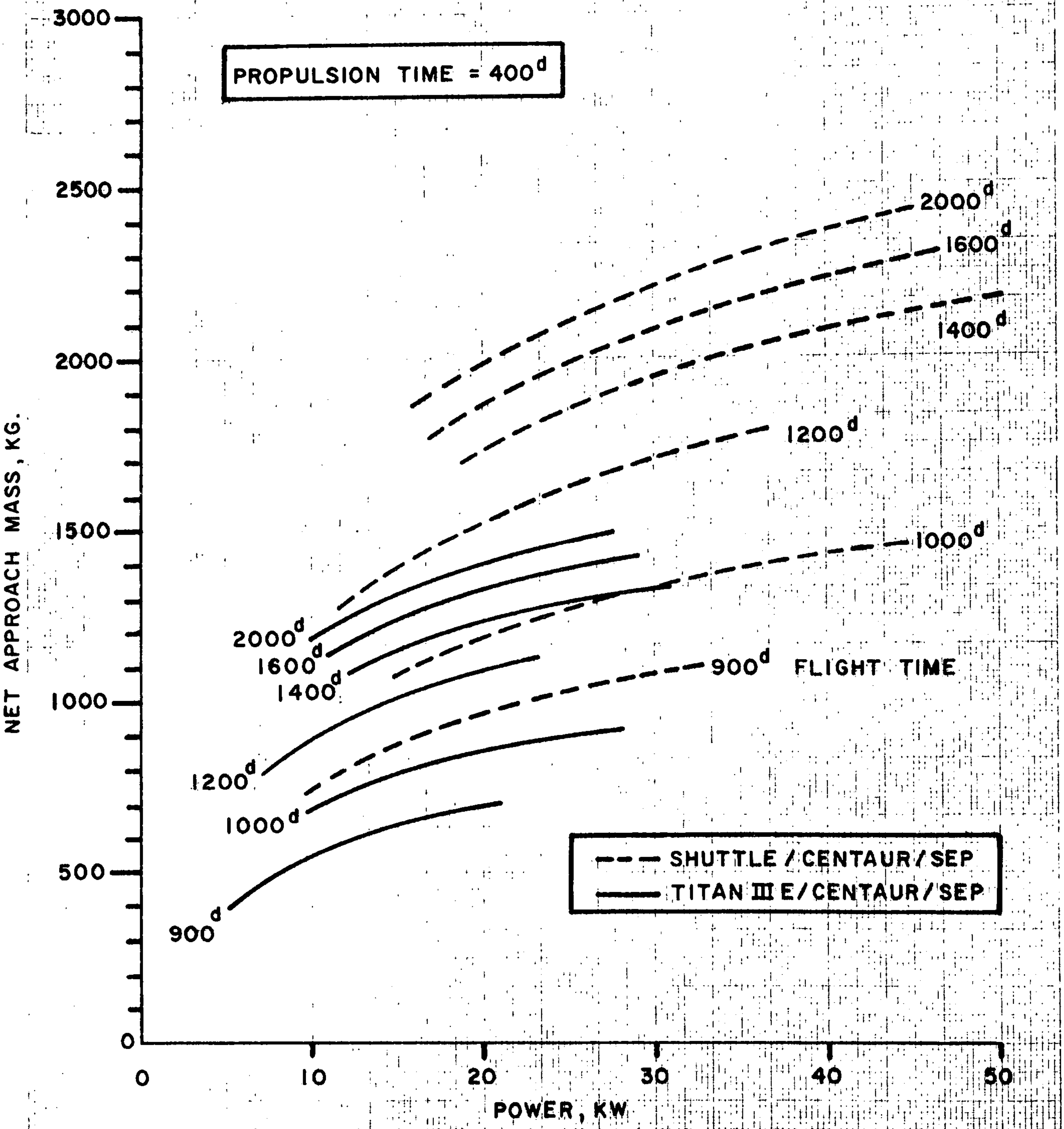


FIGURE 3. NET APPROACH MASS VS. SEP POWER FOR FIXED FLIGHT TIMES — SATURN 1980

3. PAYLOAD VERSUS FLIGHT TIME RESULTS

3.1 Jupiter Flyby Missions: 1974 - 1986

The time interval 1974 - 1986 covers more than one full cycle of Earth-Jupiter transfers. A good approximation to the performance in other years can be obtained by adding (or subtracting) 12 to the launch year. To determine the flight time for a ballistic flyby mission, a net payload equal to the spacecraft weight including midcourse propulsion system is required. For a SEP flyby, the payload must also include any provision for a stage (typically 300 kg).

Using a Titan III E/Centaur launch vehicle, with either a BII (2300) or a TE364-4 high energy upper stage, the flight time required for a payload of 750 kilograms is between 1.3 and 1.6 years. Longer than average times are needed in the years 1977 to 1980. The maximum payload which can be delivered with these Titan vehicles is generally 1000 to 1200 kg, requiring a flight time of up to two years. The exceptions are 1978 and 1979 when the payload is smaller but still more than 800 kg. The DLA constraint is apparent in the curves for 1977 to 1979 and 1984 and 1985, it causes a change in slope. It does not affect the flyby payload performance.

The introduction of the Shuttle launched Centaur/HE BII in the 1980's provides a small (less than 100 days) improvement in the flight time for a nominal 750 kg flyby spacecraft. The maximum payload increases to 200 kg (in 1981).

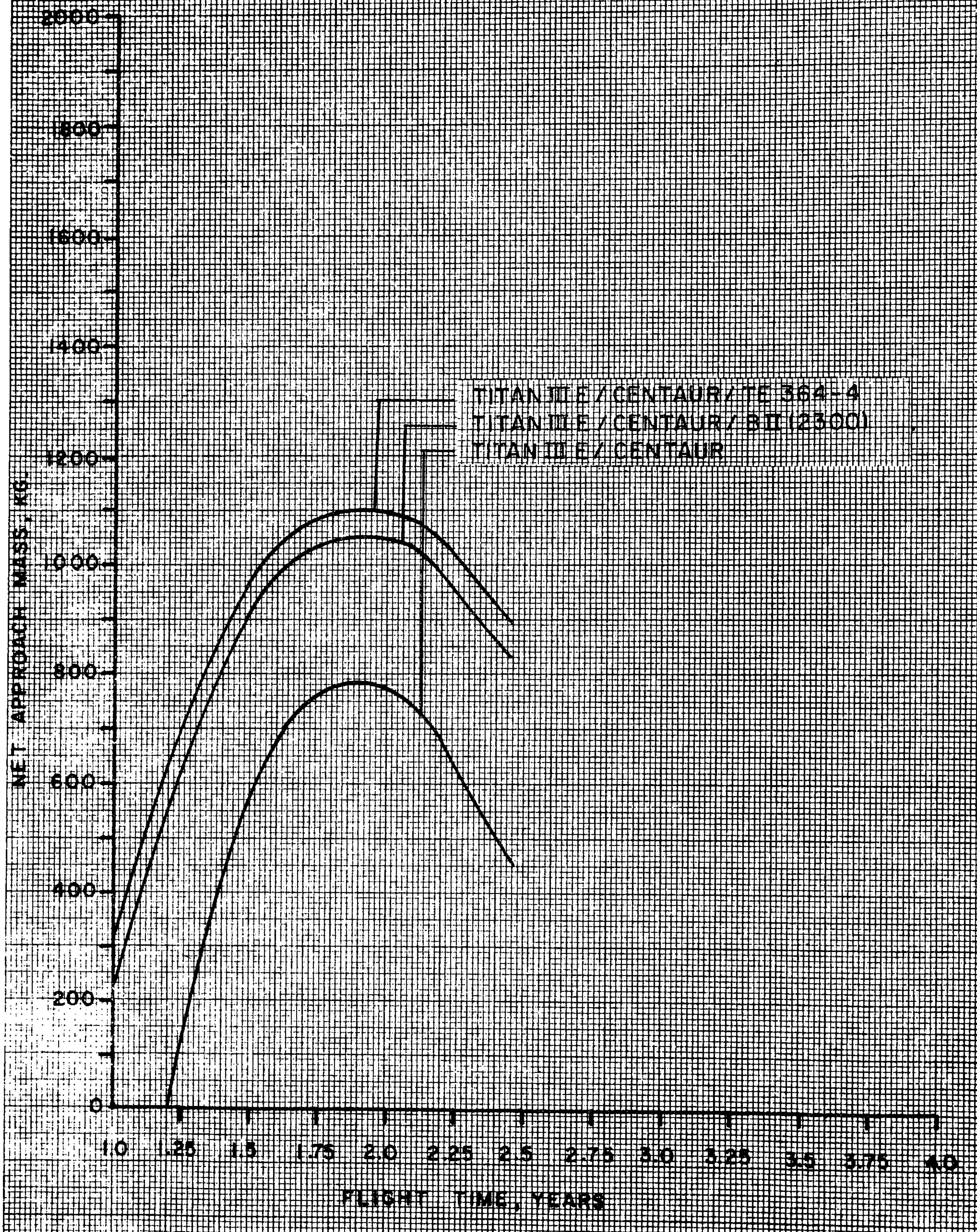


FIGURE 4. 1974 JUPITER FLYBY

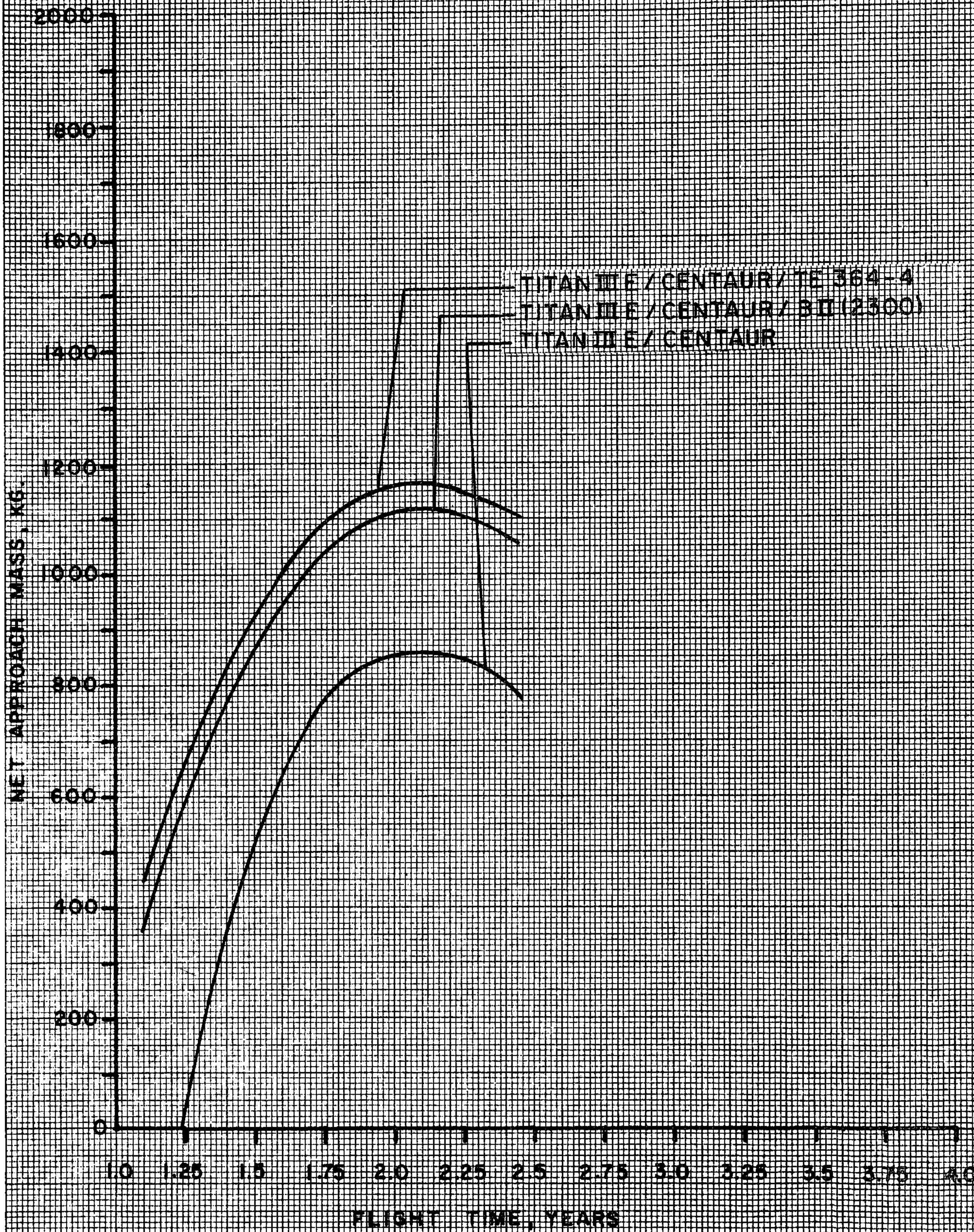


FIGURE 5. 1975 JUPITER FLYBY

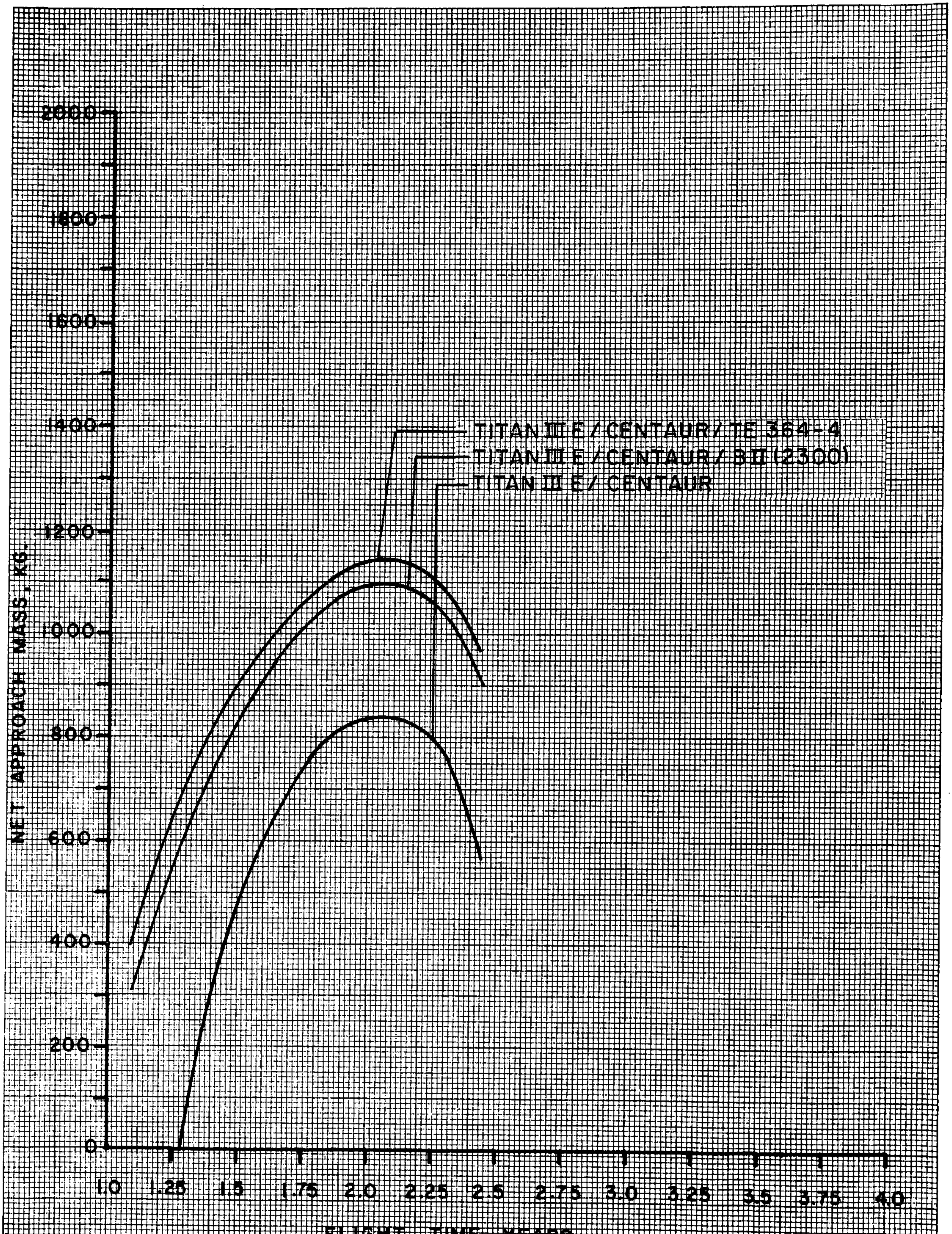


FIGURE 6. 1976 JUPITER FLYBY

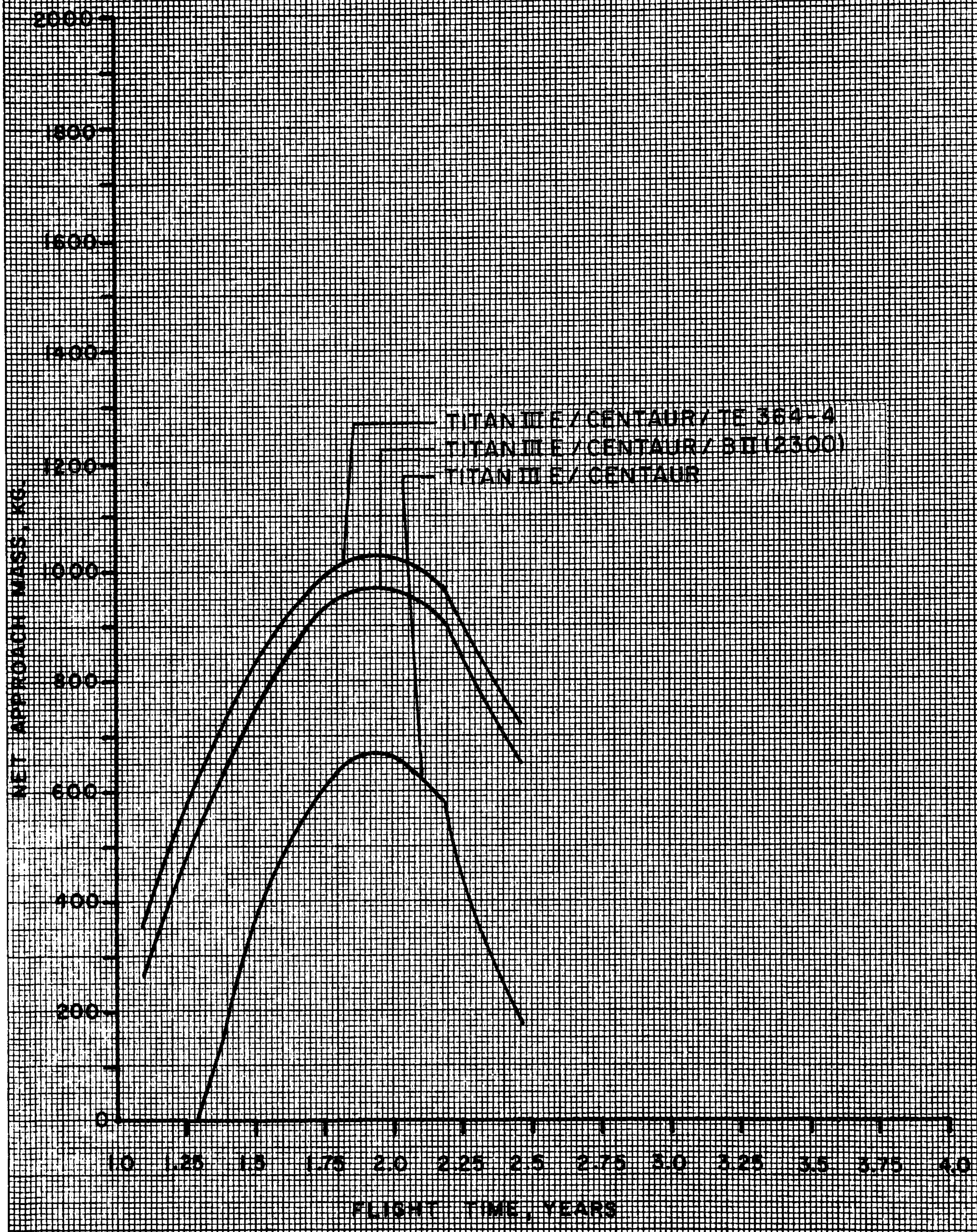


FIGURE 7. 1977 JUPITER FLYBY

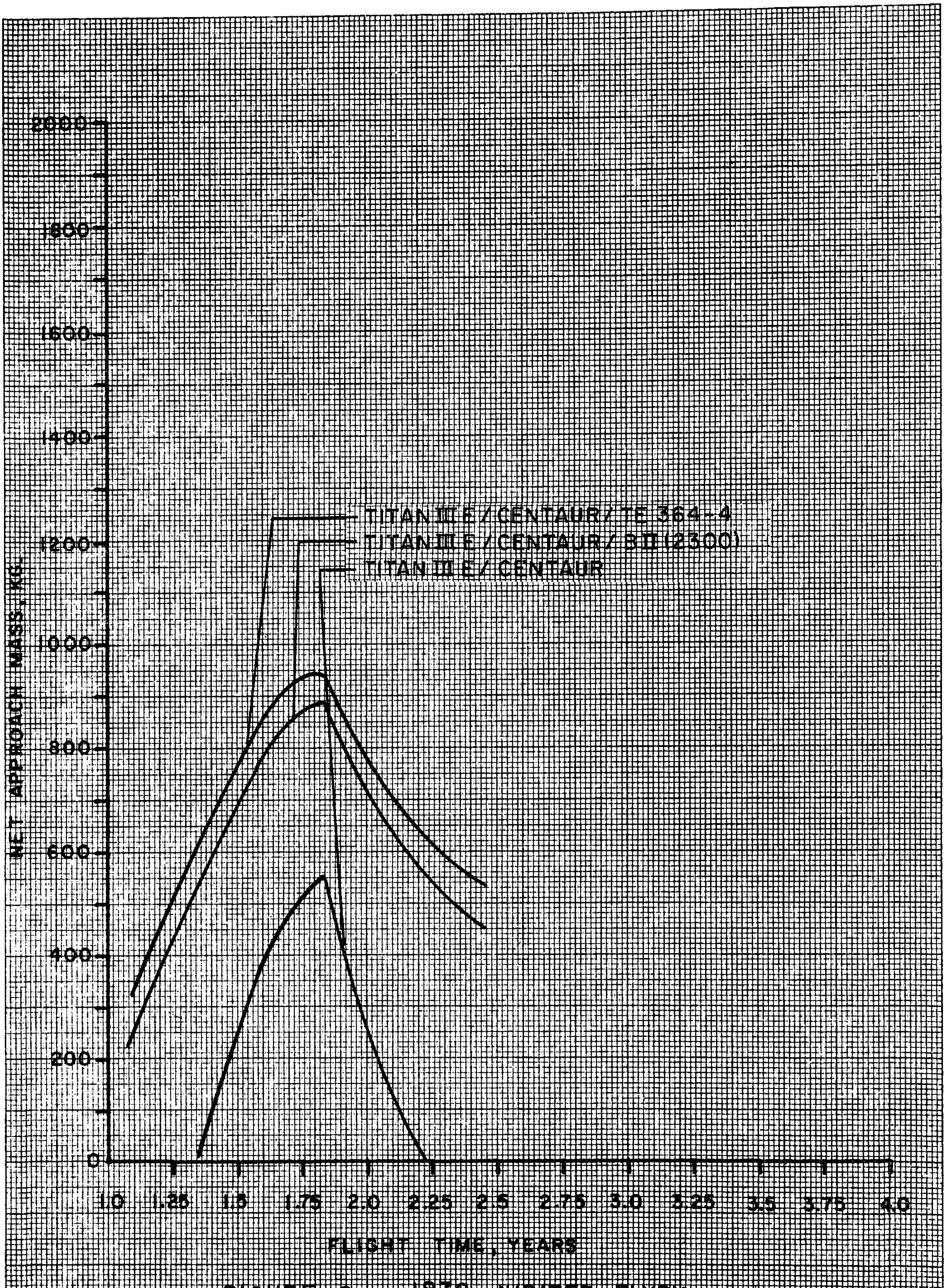


FIGURE 8. 1978 JUPITER FLYBY

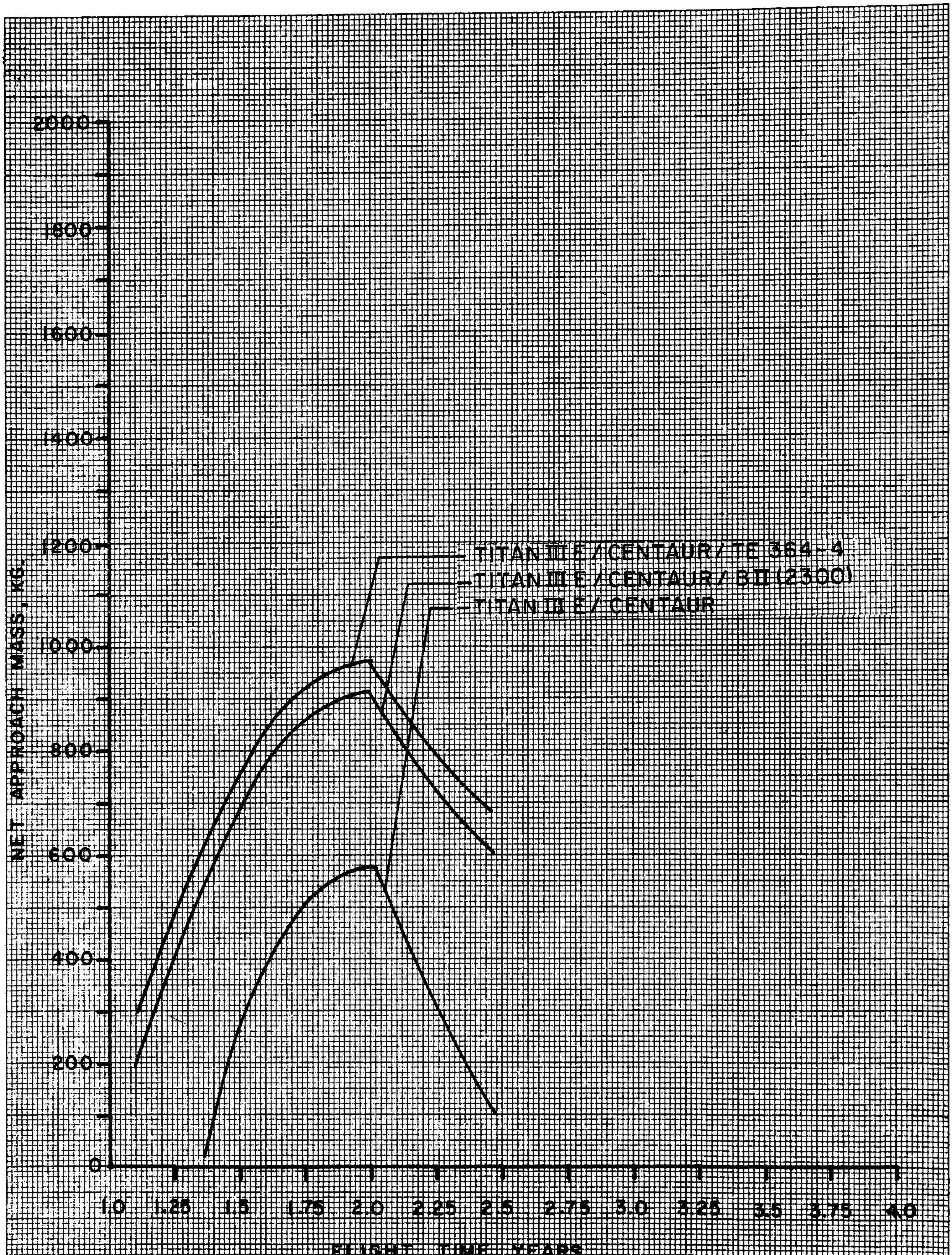


FIGURE 9. 1979 JUPITER FLYBY

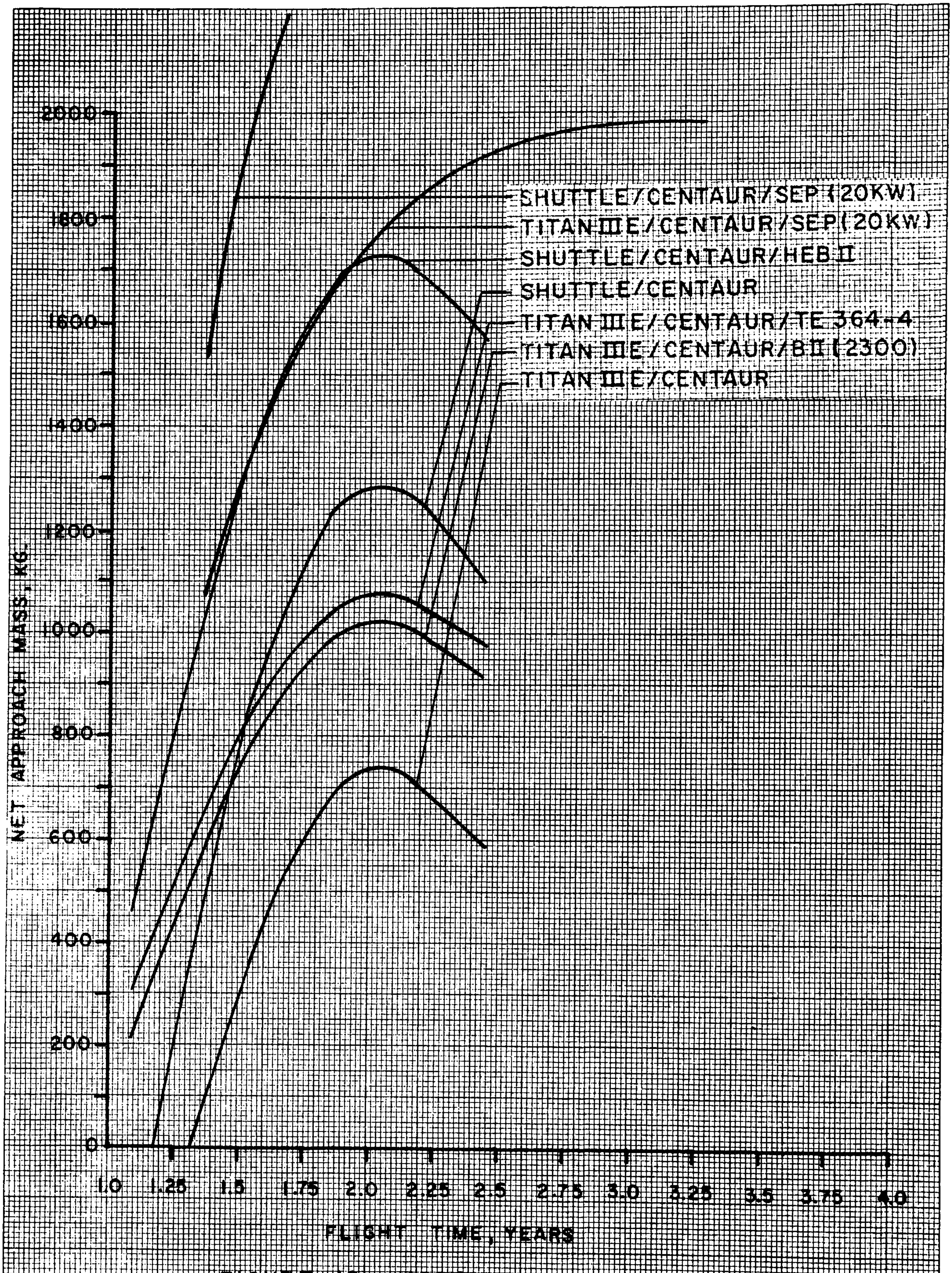


FIGURE 10. 1980 JUPITER FLYBY

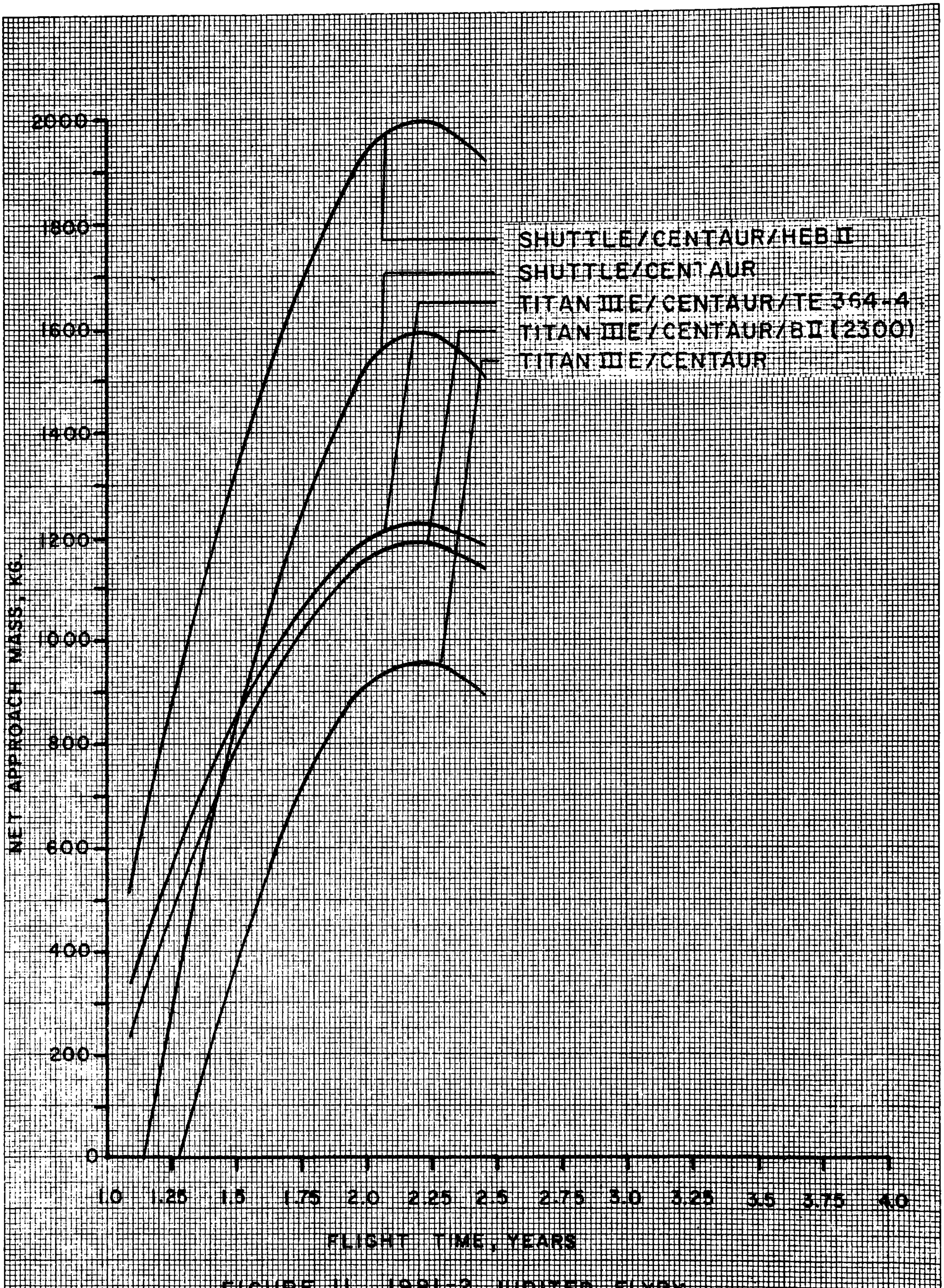


FIGURE 11. 1981-2 JUPITER FLYBY

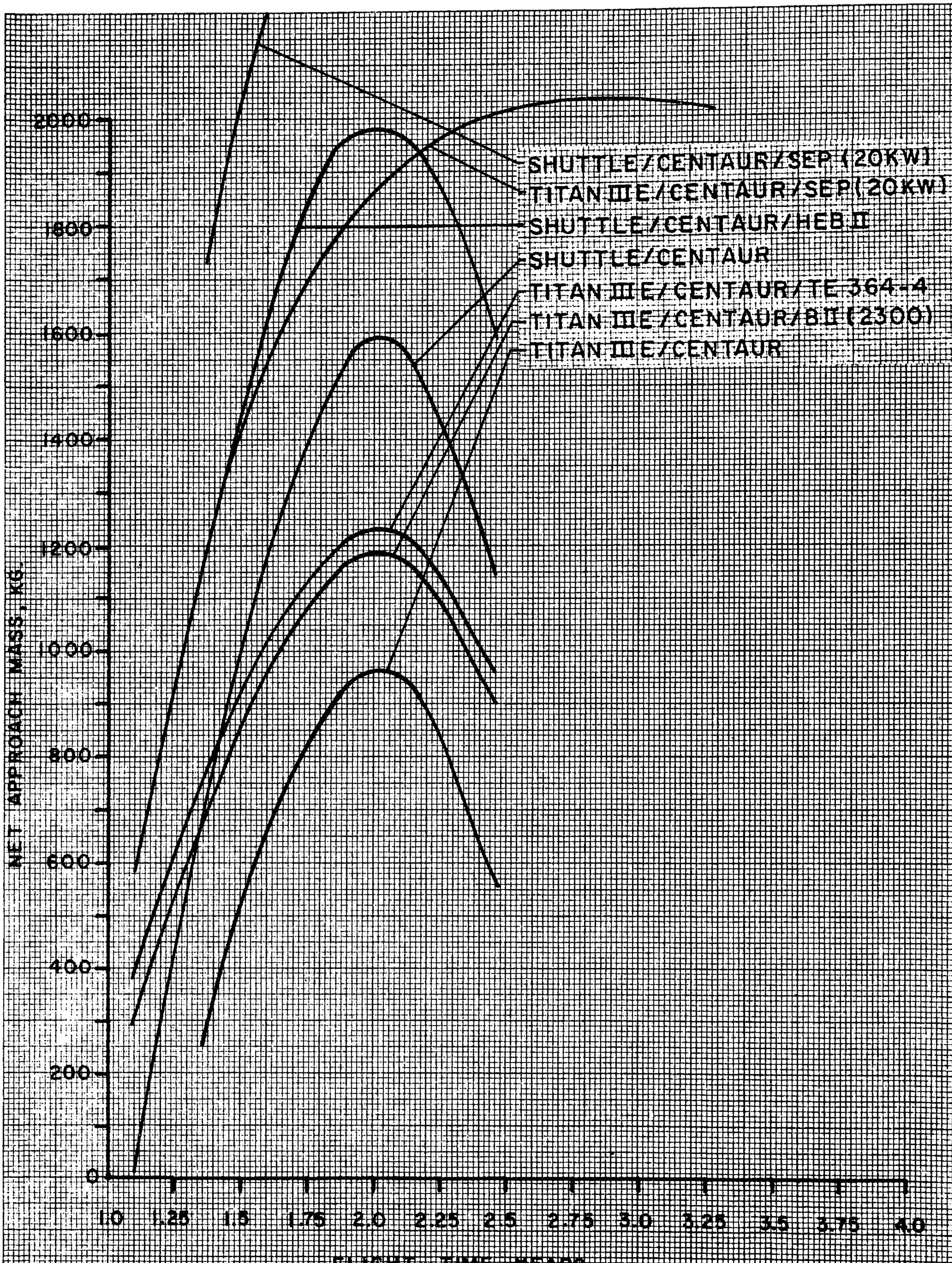


FIGURE 12. 1983 JUPITER FLYBY

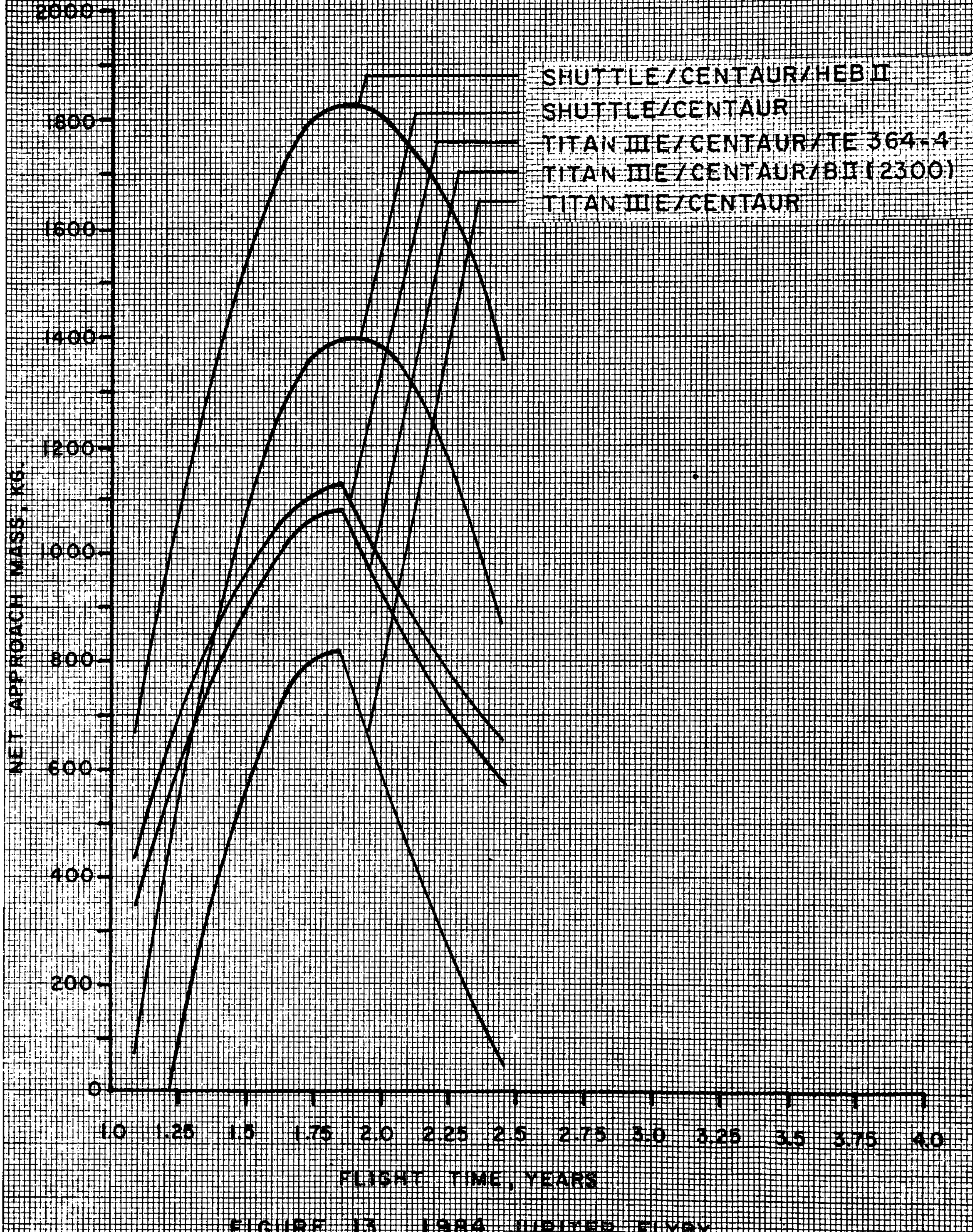


FIGURE 13. 1984 JUPITER FLYBY

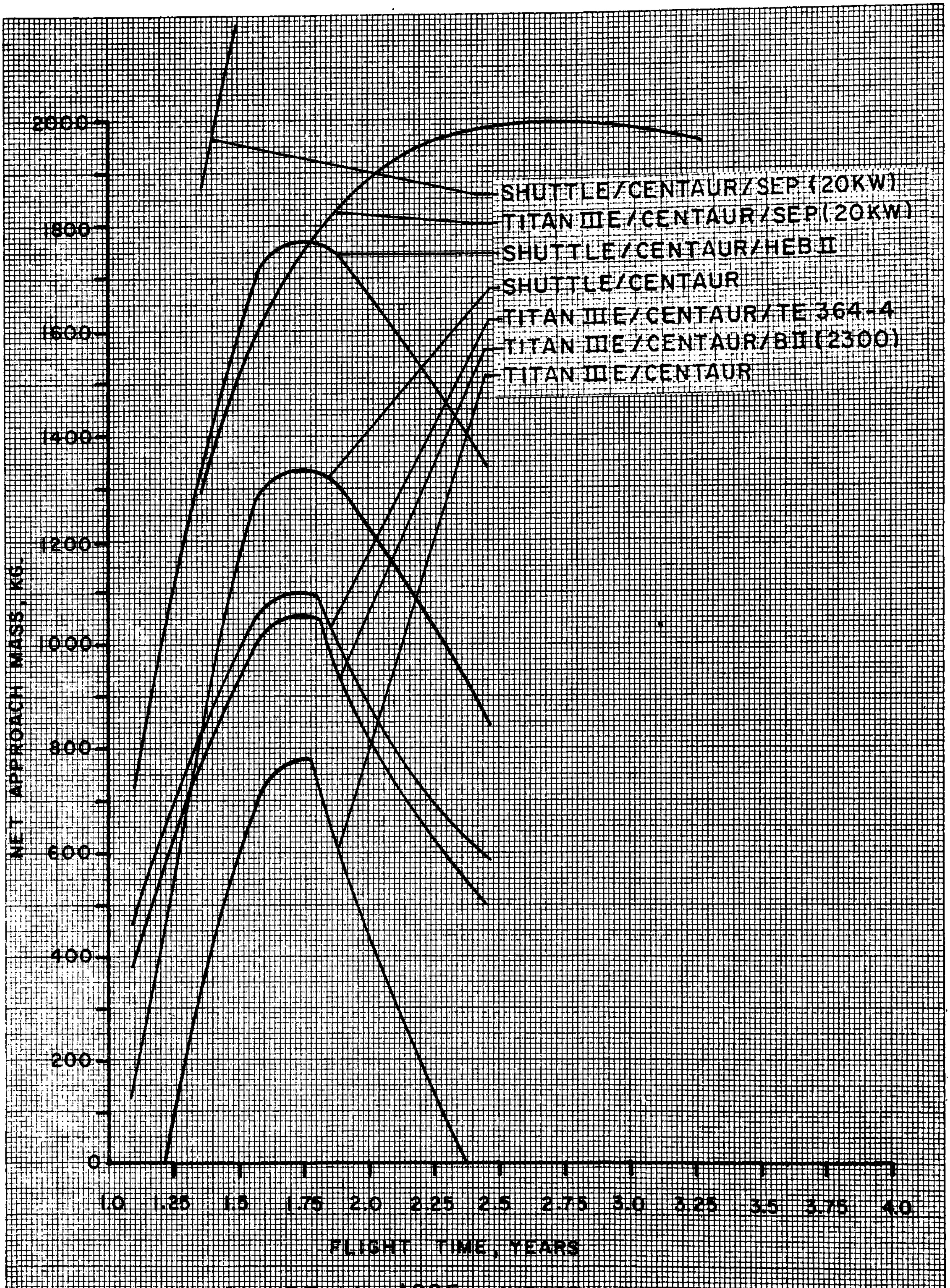


FIGURE 14. 1985 JUPITER FLYBY

c

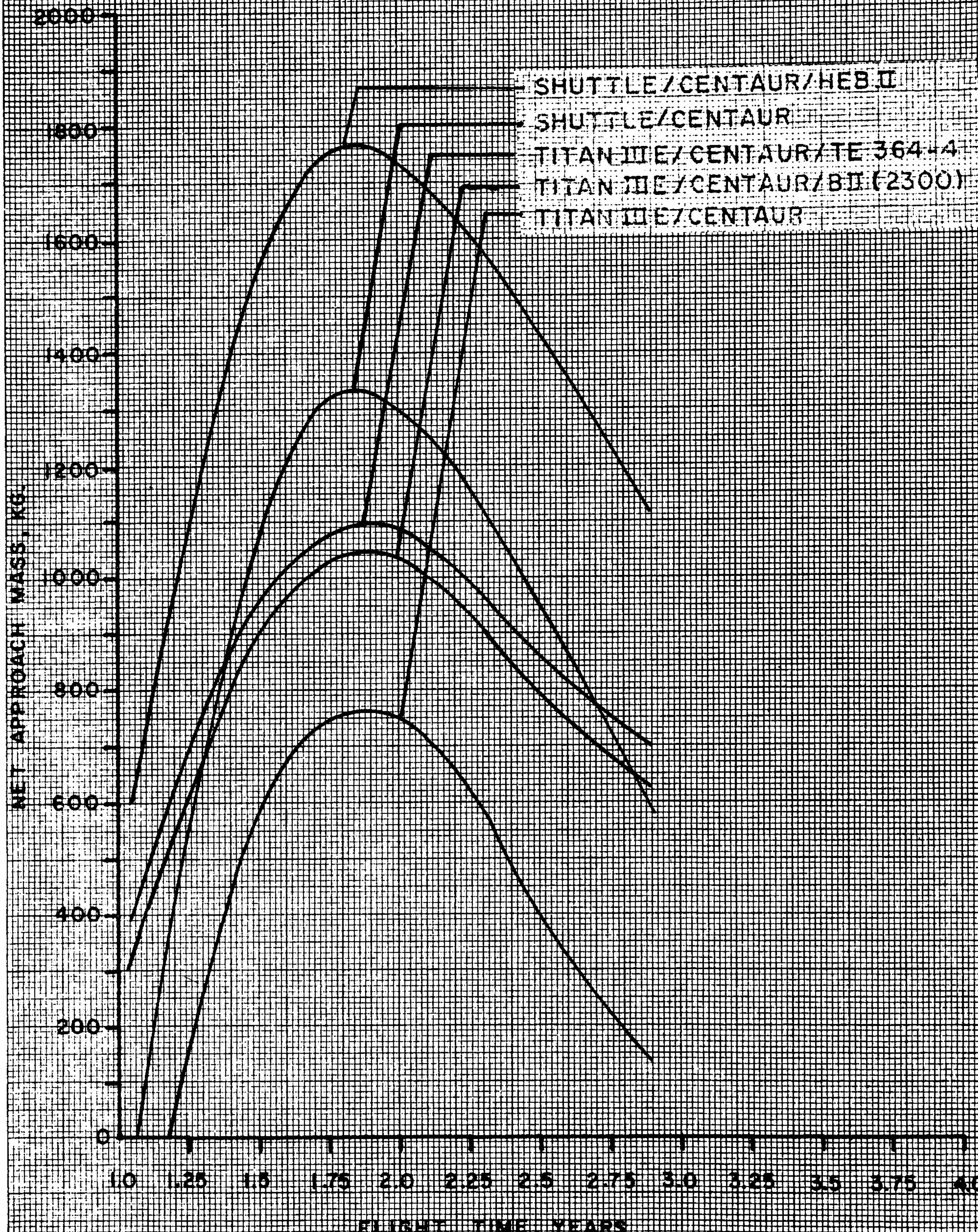


FIGURE 15. 1986 JUPITER FLYBY

3.2 Jupiter Orbiter Missions: 1976 to 1986

The time interval 1976 to 1986 covers almost one full cycle of Earth-Jupiter transfers. For a good approximation to the performance in other years add or subtract 12 to the launch year. The payload results reflect a provision of 250 m/sec for midcourse trajectory corrections and orbit trim maneuvers. The SEP curves assume that a 300 kg stage was jettisoned prior to orbit capture.

The orbits considered have a periapse radius of 4 Jupiter radii and periods of 60, 30 and 15 days. Orbiter missions in the late 1970's are launched by a Titan III E/Centaur/BII and use an earth-storable retro propulsion system. None has a net mass in orbit exceeding 600 kg. In 1978 and 1979 the DLA constraint causes a reduction in maximum orbiter payload of 17 and 5 percent, respectively. Changing to space-storable propellants in the 1980's yields some increase in the payloads but does not permit a 750 kg spacecraft to be placed in these orbits. However, the Shuttle/Centaur/HE BII is capable of delivering this nominal spacecraft with a flight time of 1.7 to 1.9 years for a 30 day period orbit. This mission can also be accomplished by adding a 20 kw SEP system to the Titan III E/Centaur, although the flight time is about 75 days longer. A Shuttle launched 20 kw SEP stage could place over 1000 kg in the orbits considered.

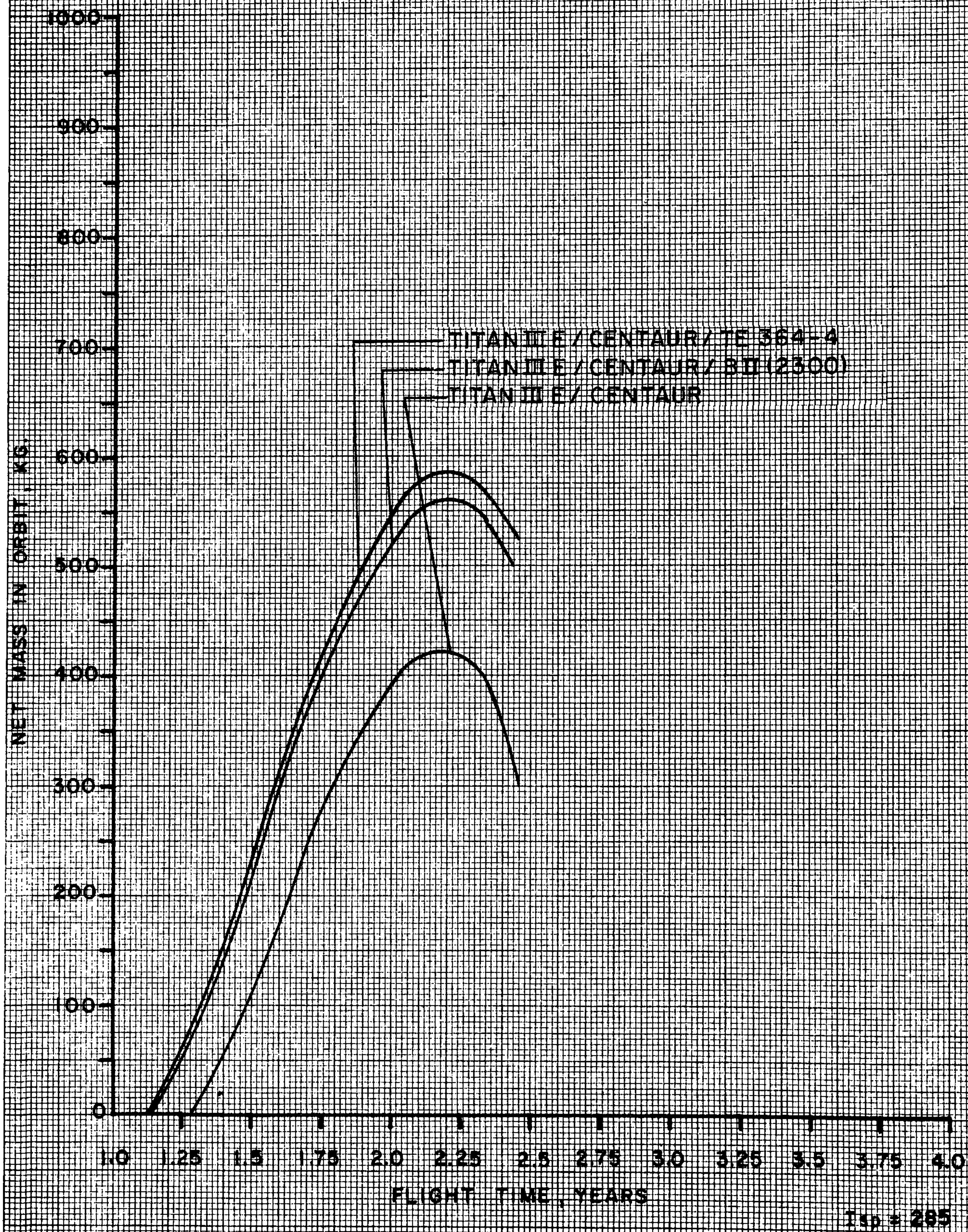


FIGURE 16. 1976 JUPITER 60[d] ORBITER

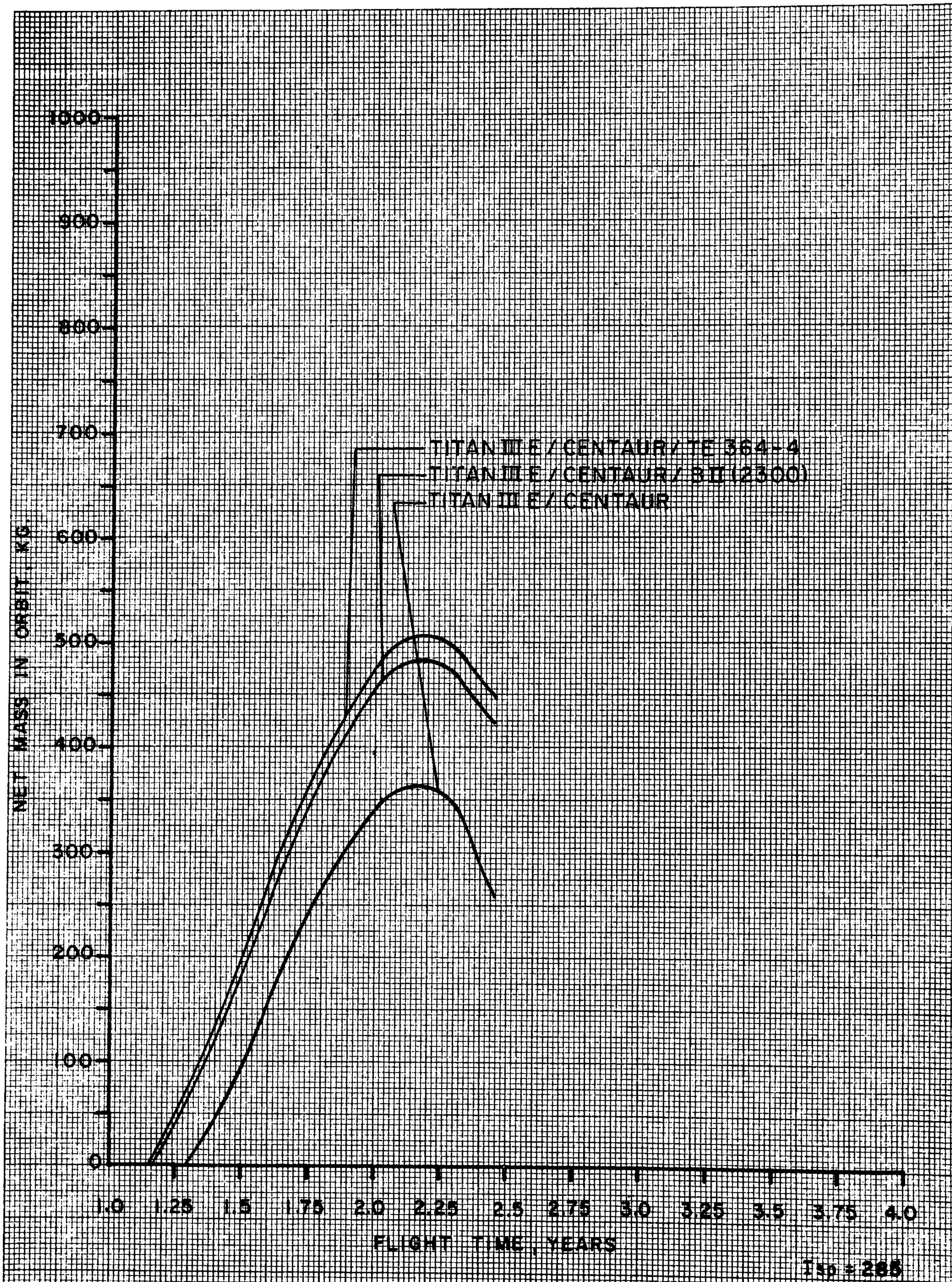


FIGURE 17. 1976 JUPITER 30° ORBITER

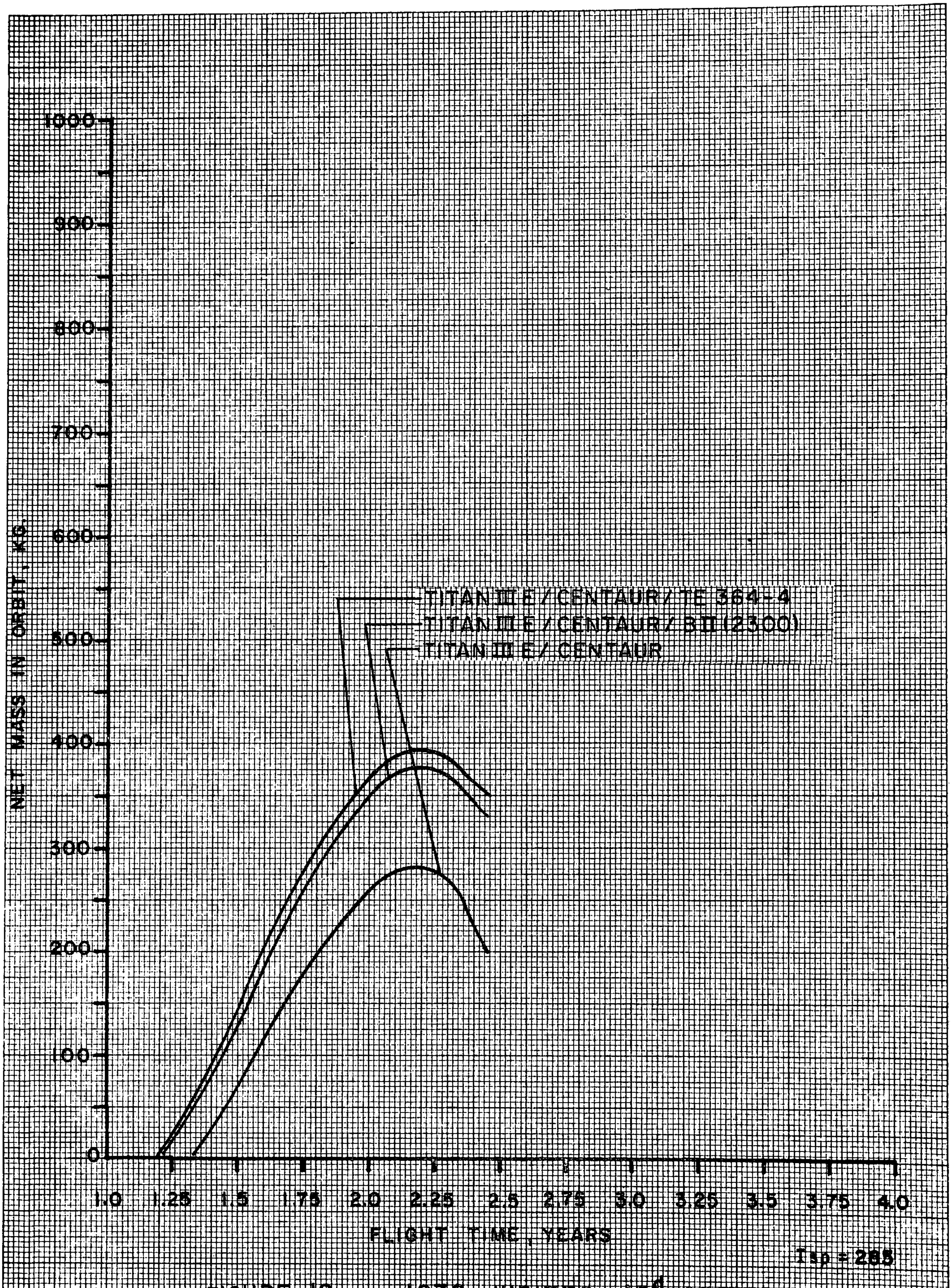


FIGURE 18. 1976 JUPITER 15^d ORBITER

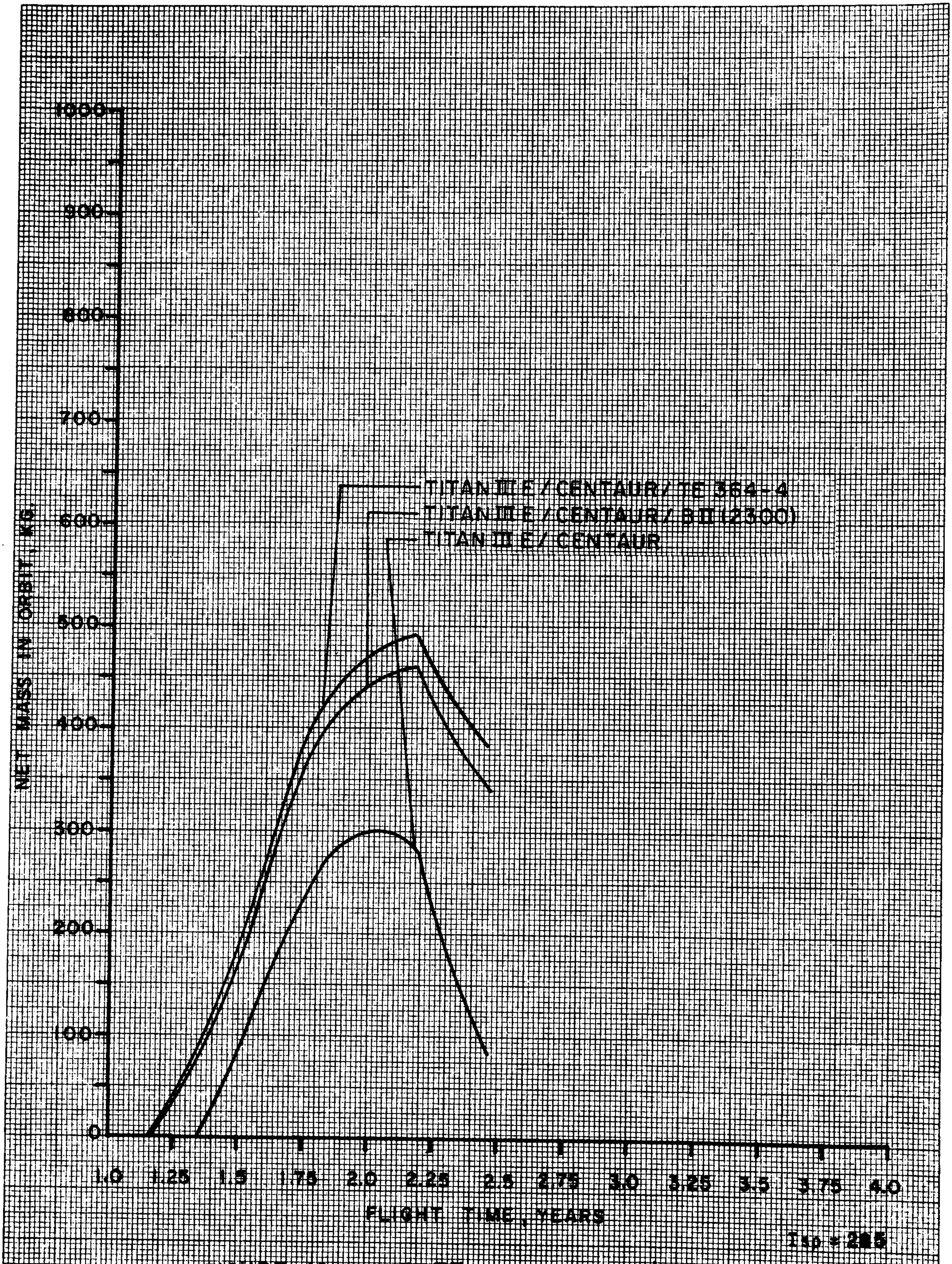


FIGURE 19. 1977 JUPITER 60° ORBITER

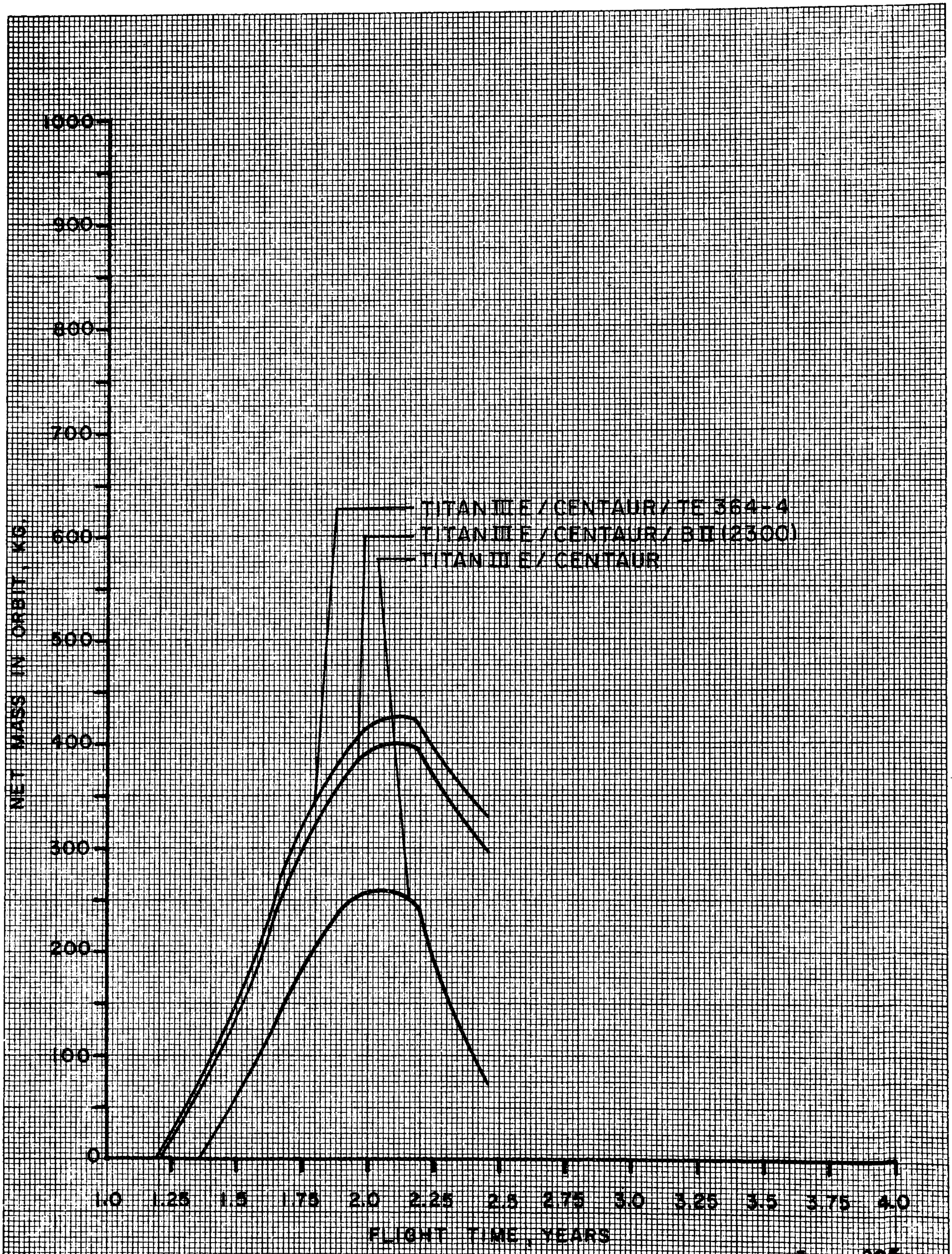


FIGURE 20. 1977 JUPITER 30^d ORBITER

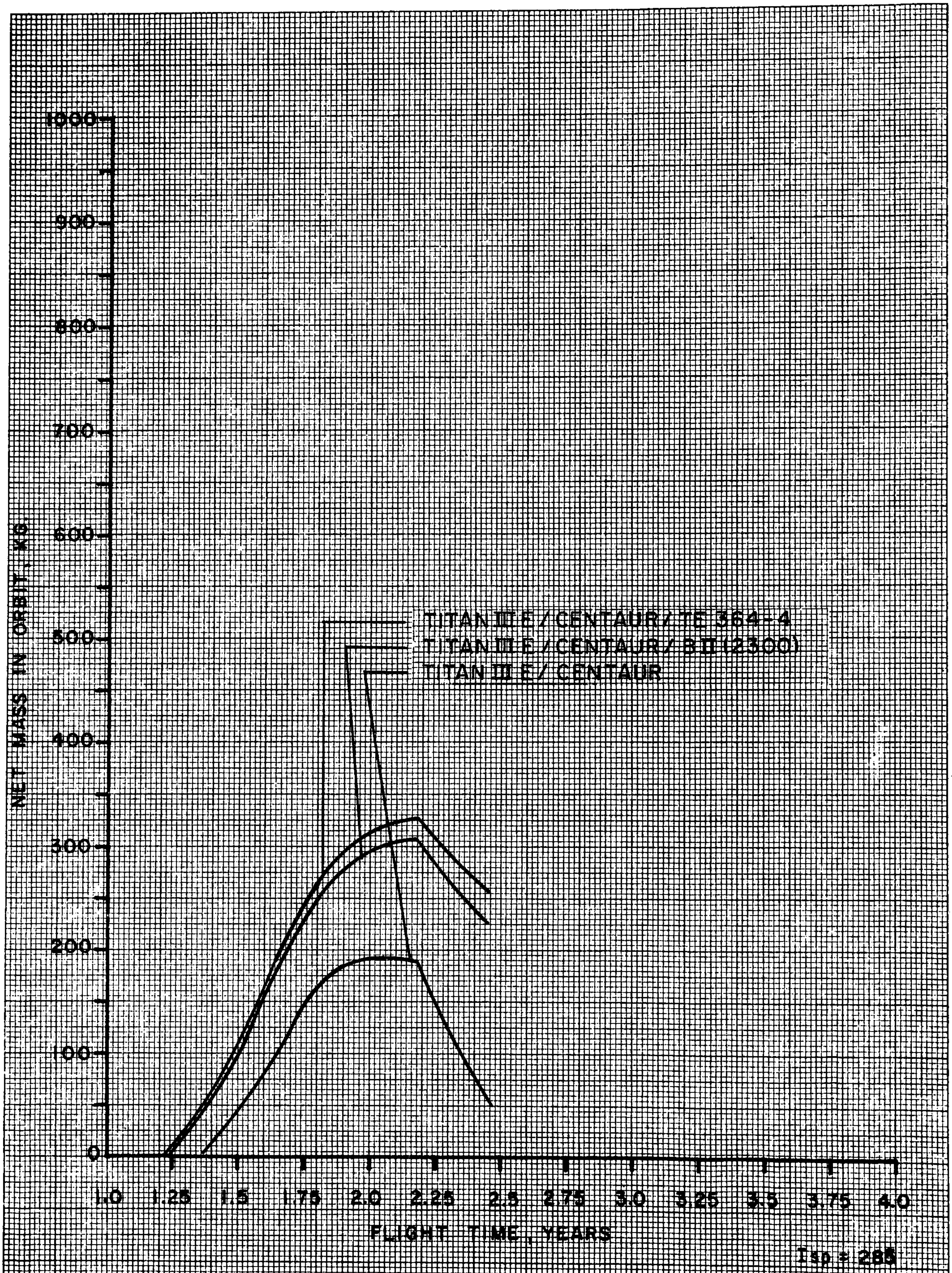


FIGURE 21. 1977 JUPITER 15d ORBITER

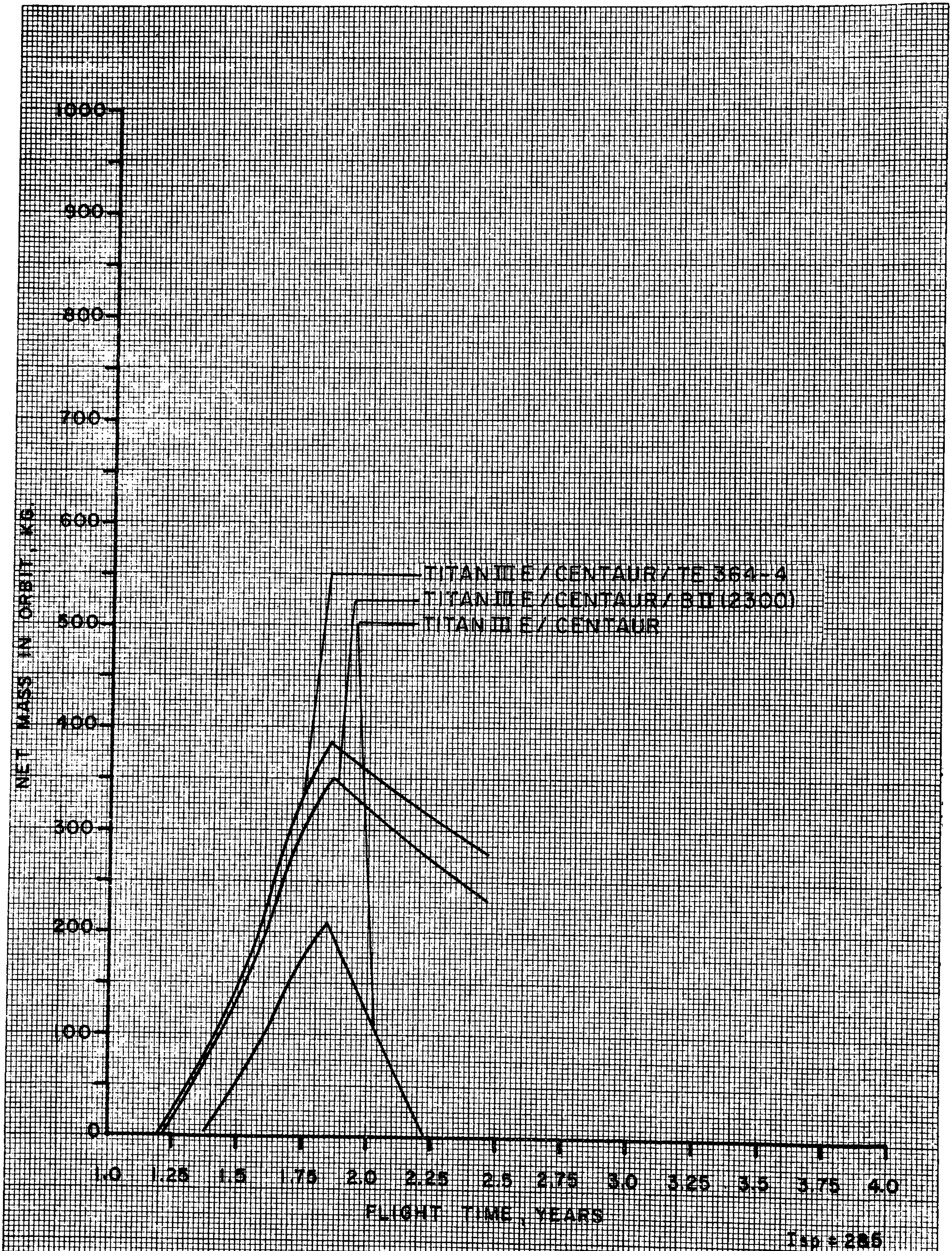


FIGURE 22. 1978 JUPITER 60° ORBITER

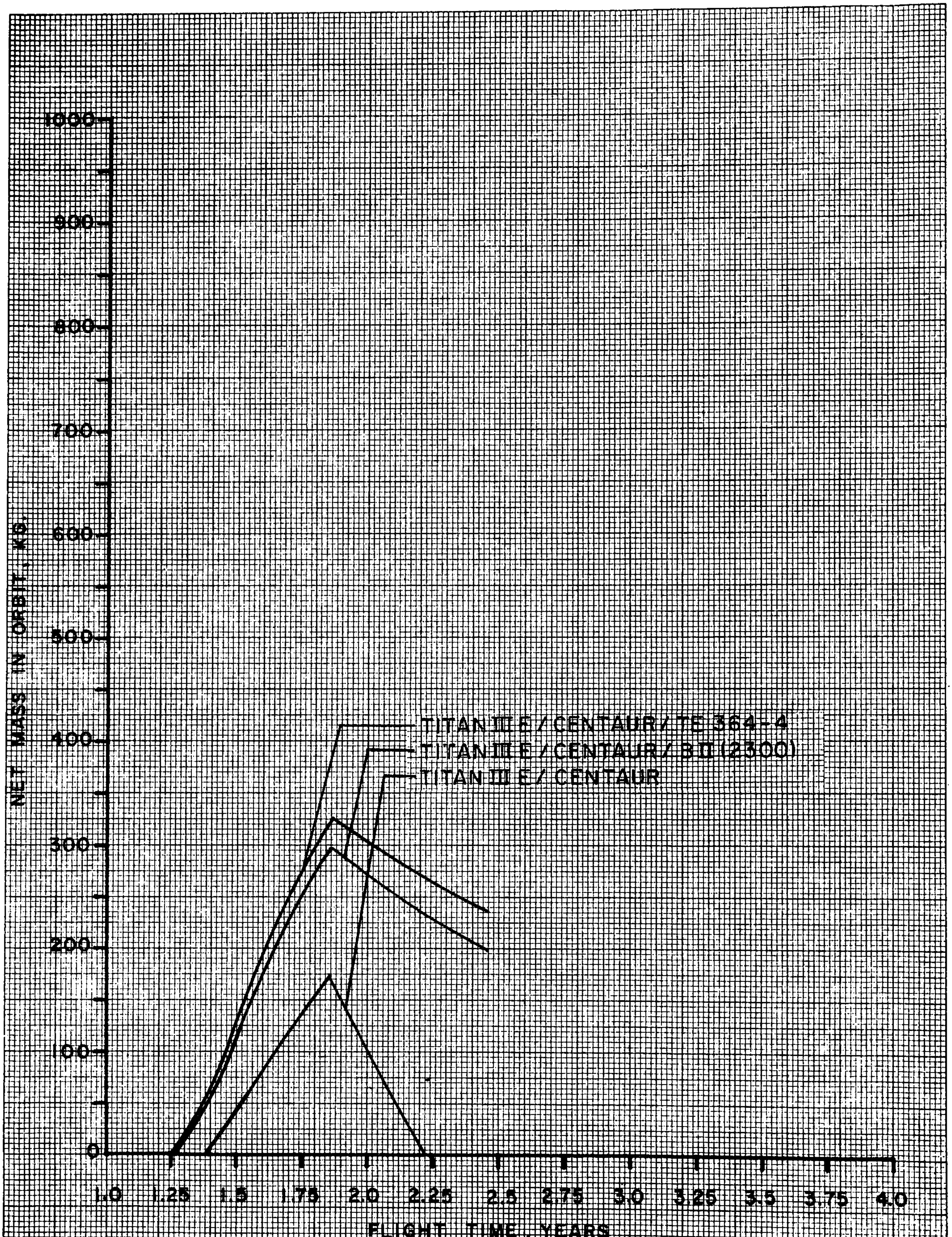


FIGURE 23. 1978 JUPITER 30d ORBITER

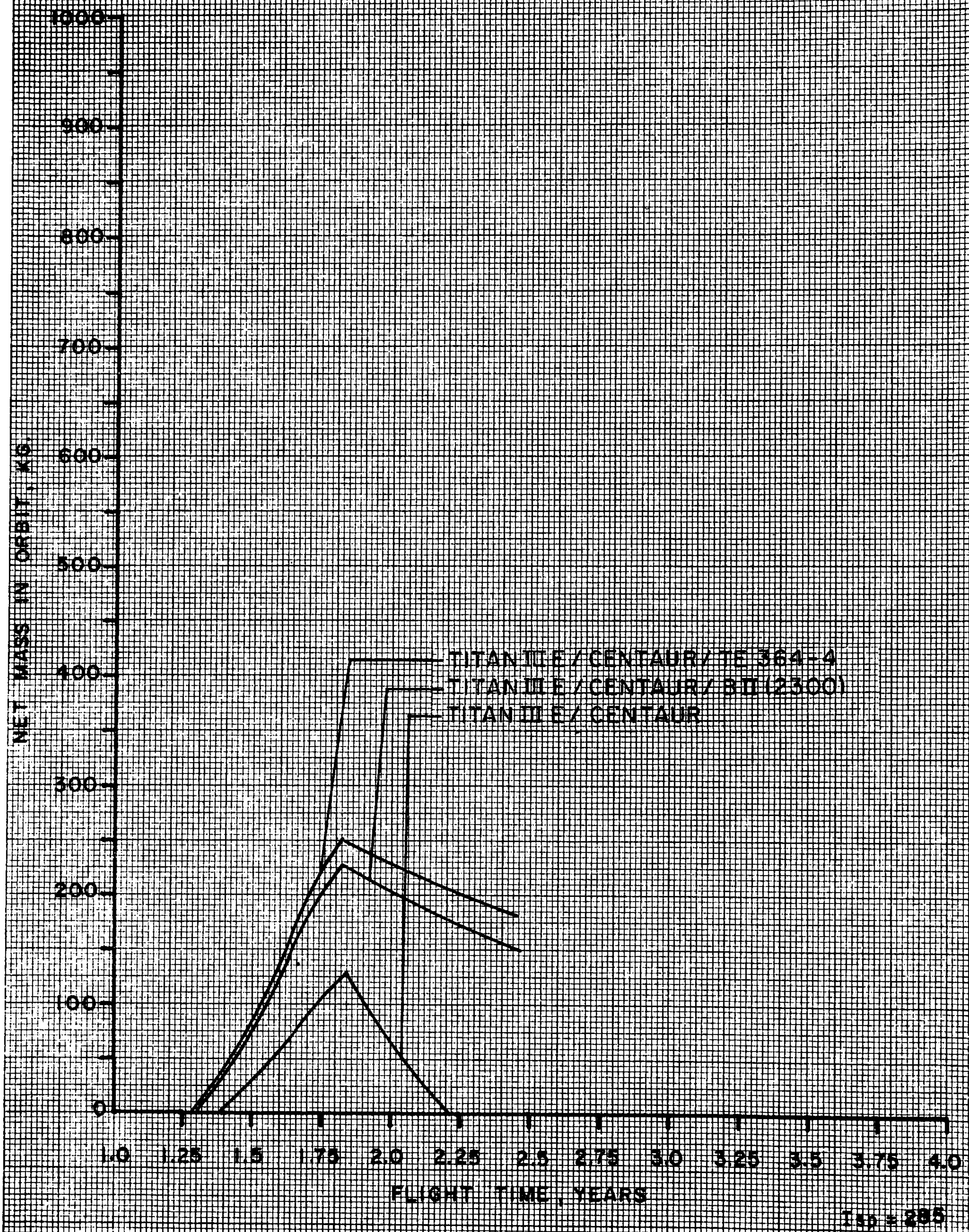


FIGURE 24. 1978 JUPITER 15° ORBITER

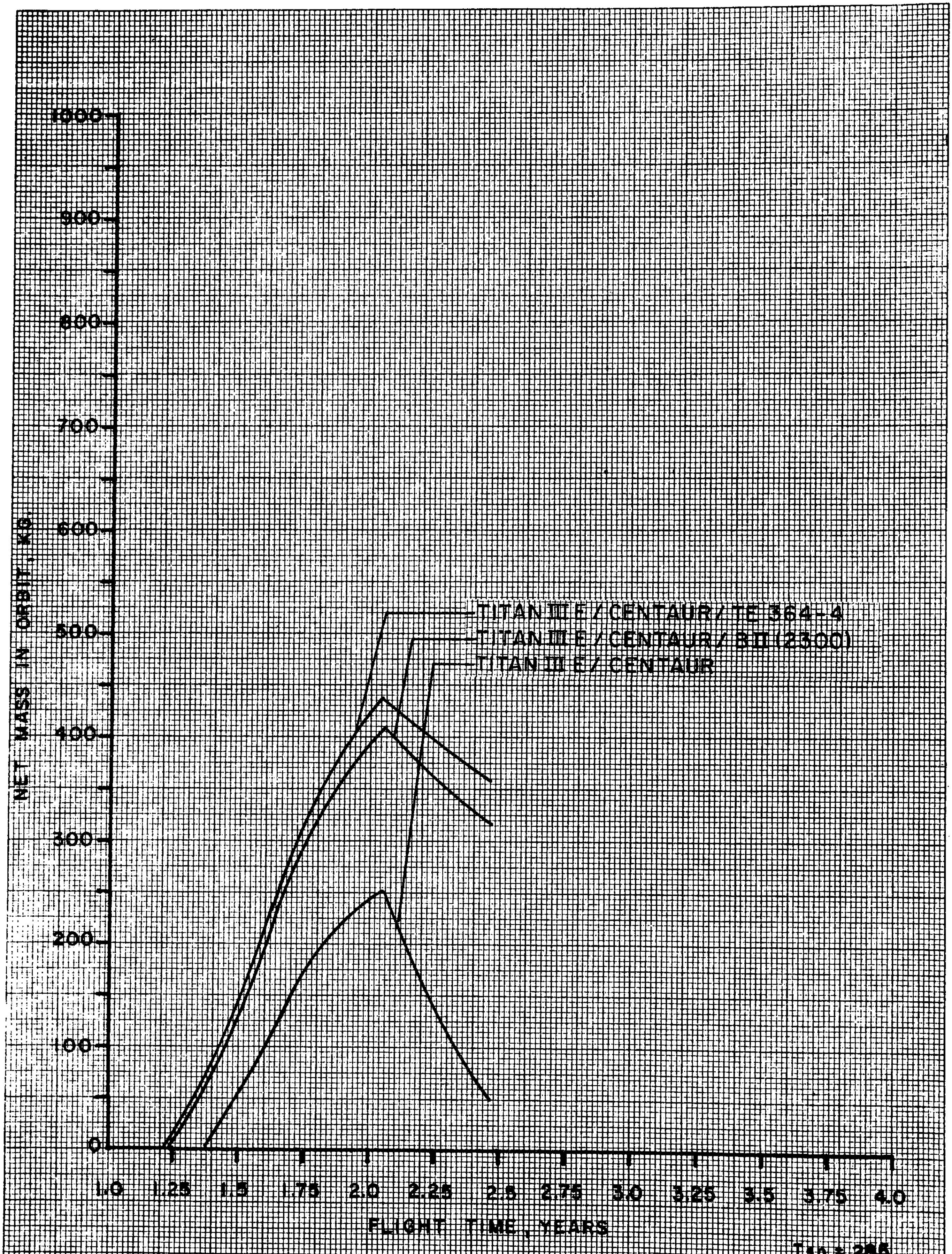


FIGURE 25. 1979 JUPITER 60° ORBITER

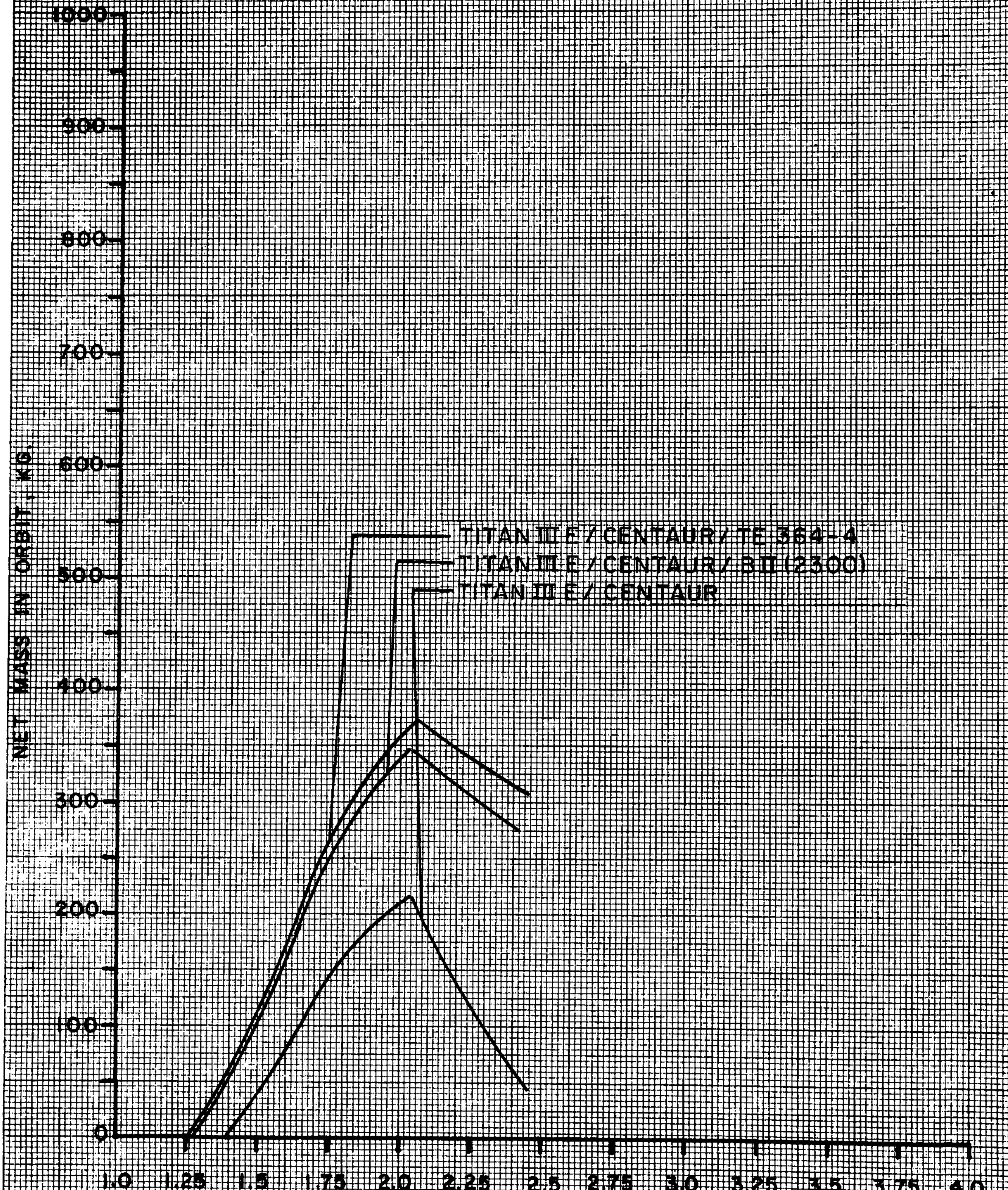


FIGURE 26. 1979 JUPITER 30d ORBITER

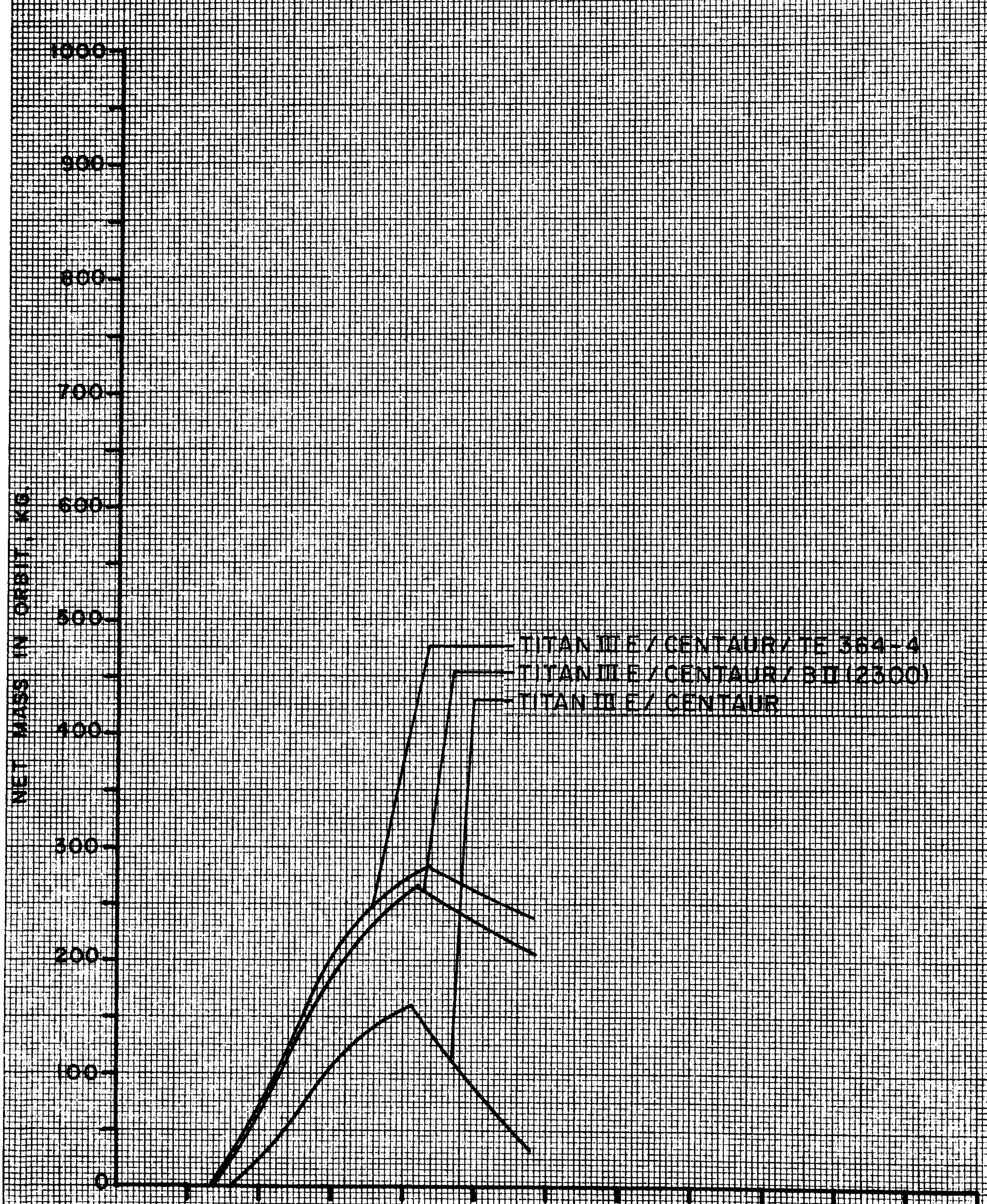


FIGURE 27. 1979 JUPITER 15[d] ORBITER

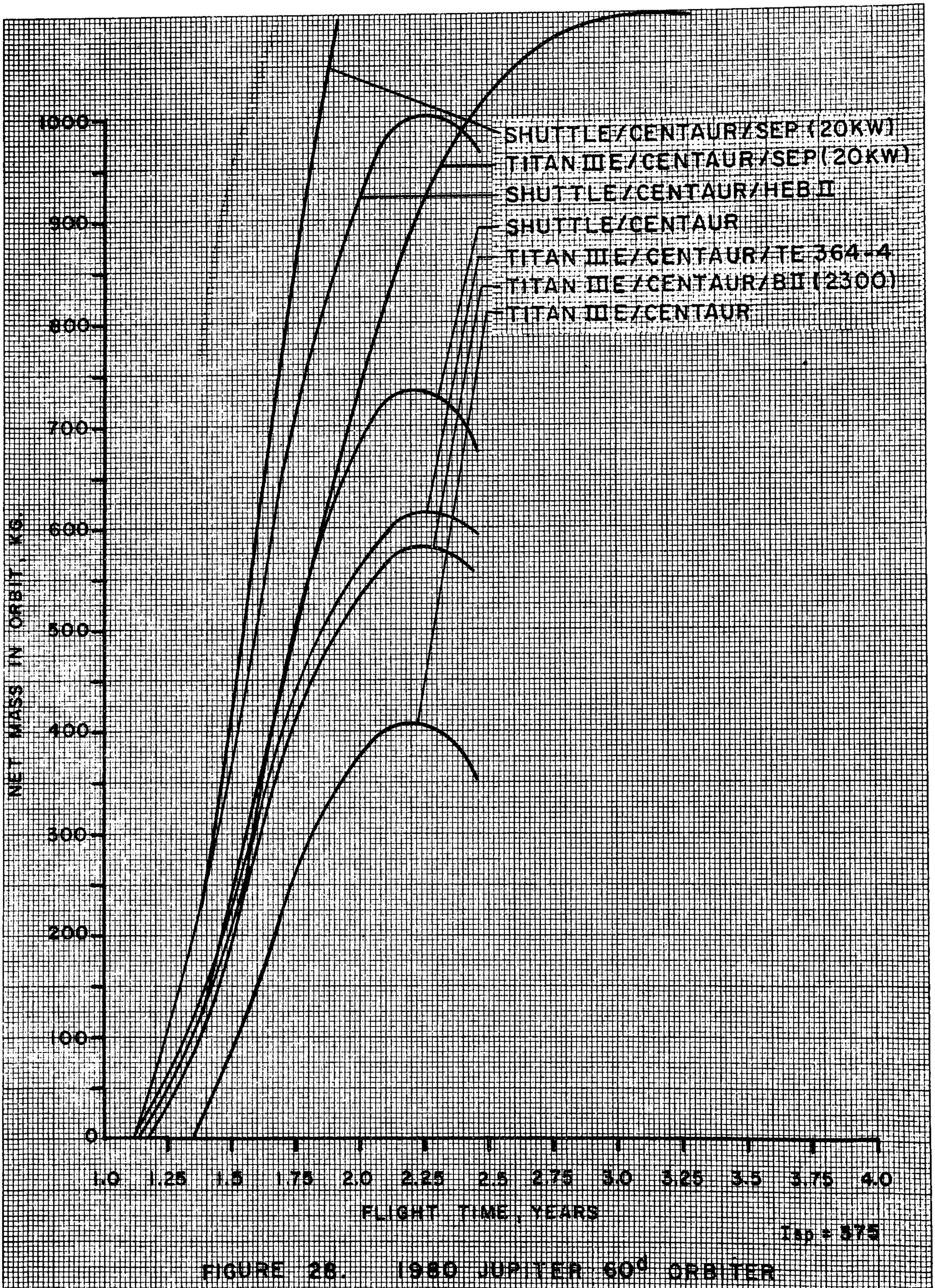


FIGURE 28. 1980 JUPITER 60^d ORBITER

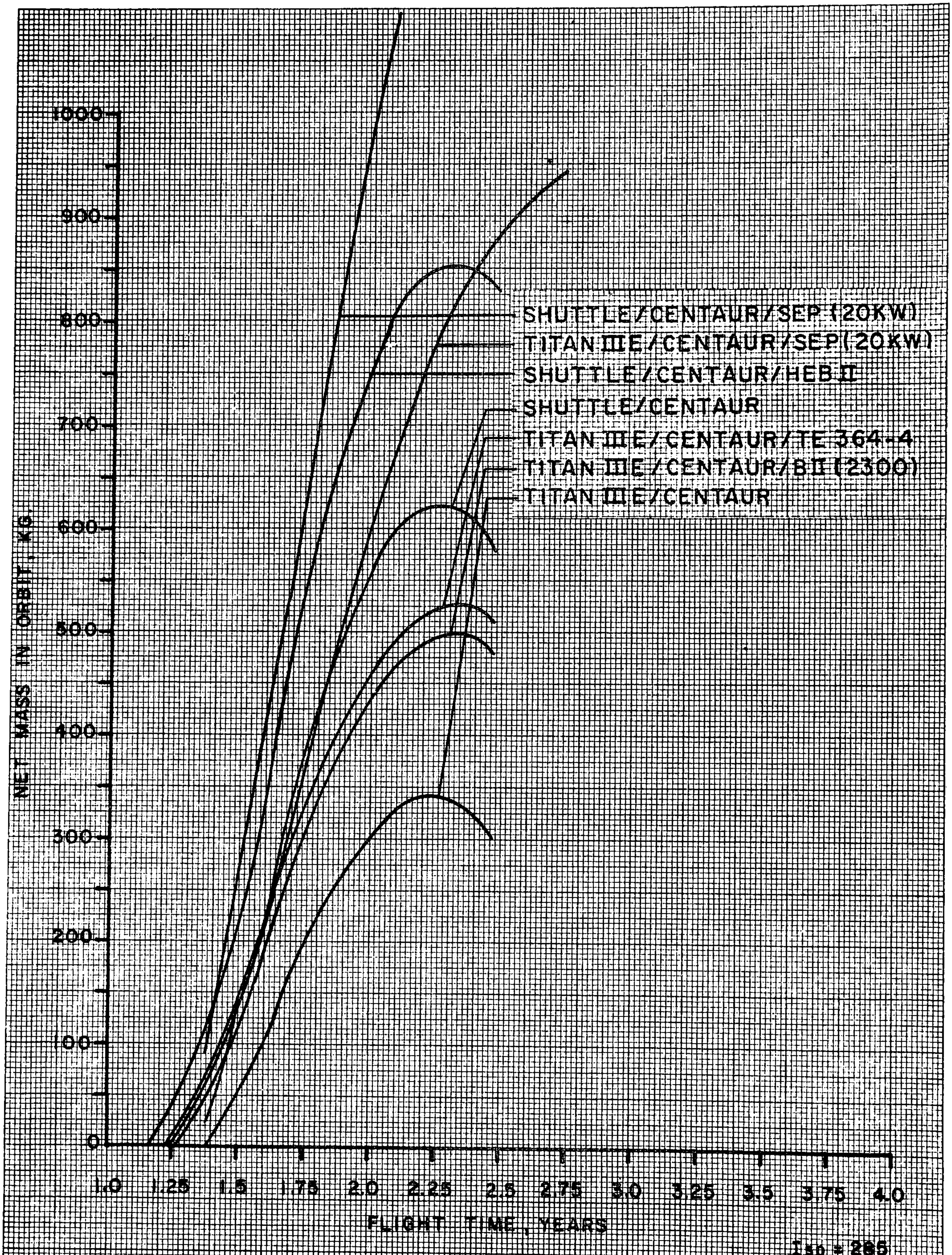


FIGURE 29. 1980 JUPITER 60^d ORBITER

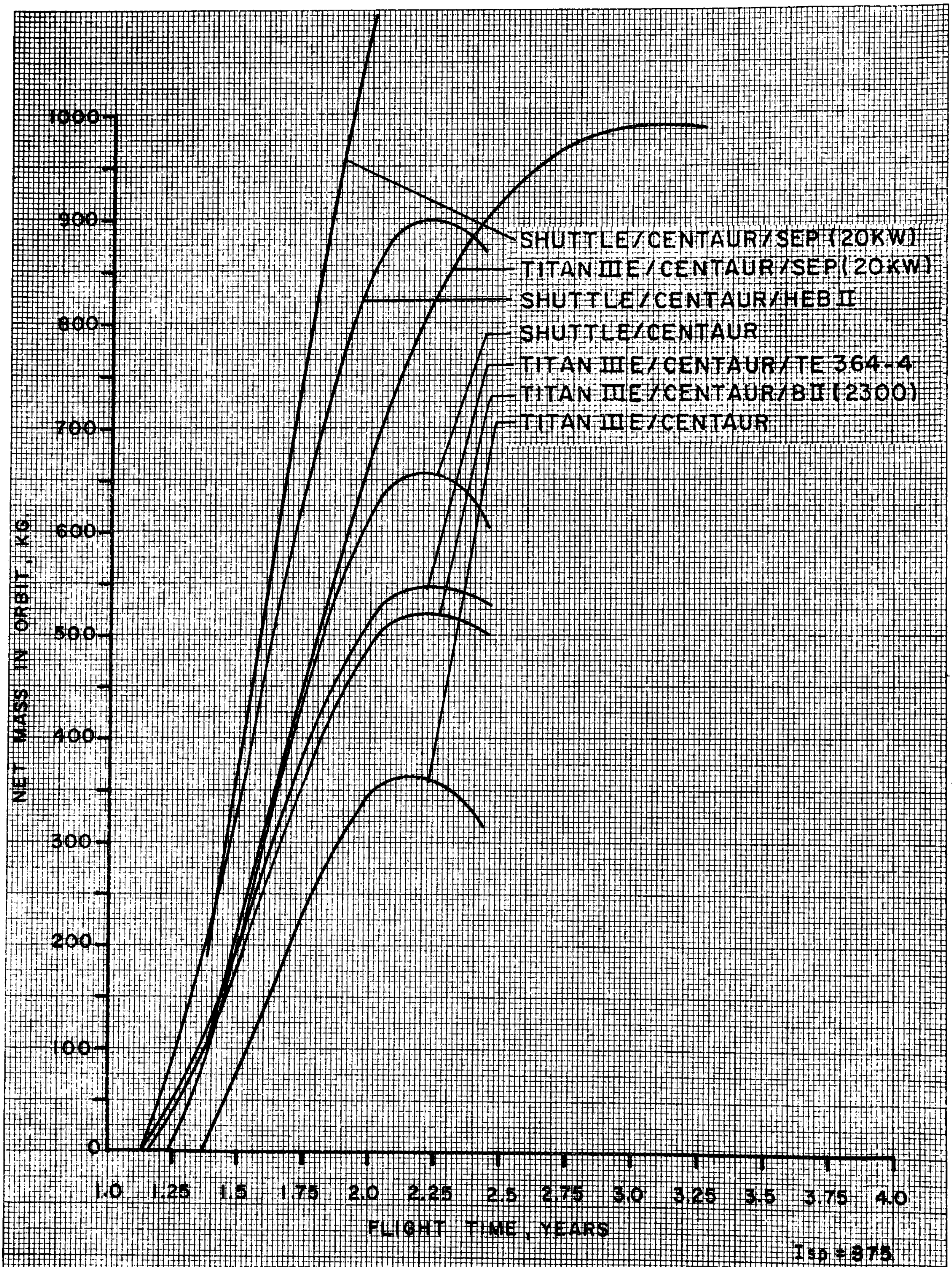


FIGURE 30. 1980 JUPITER 30[d] ORBITER

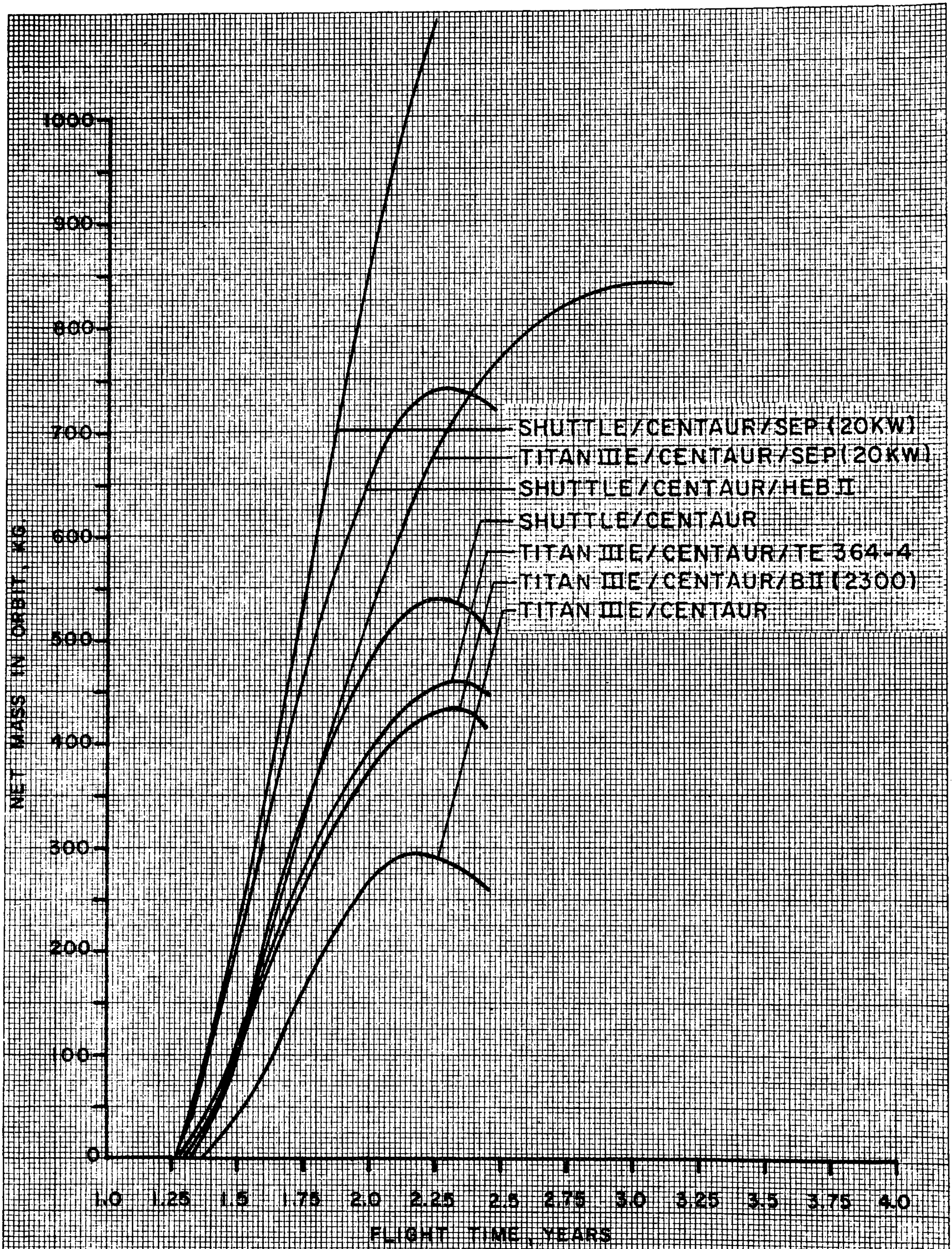


FIGURE 31. 1980 JUPITER 30^{d} ORBITER

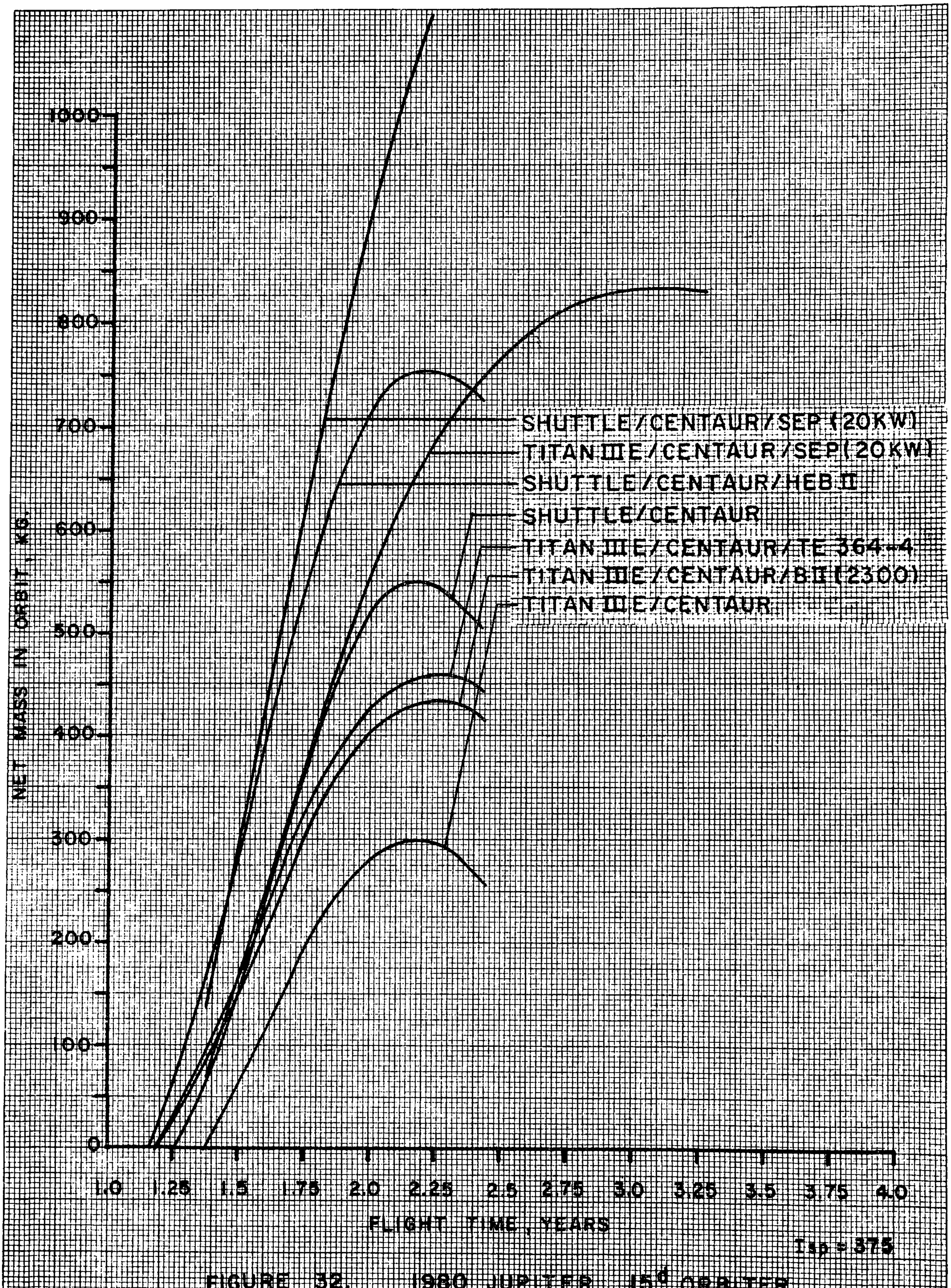


FIGURE 32. 1980 JUPITER 15^d ORBITER

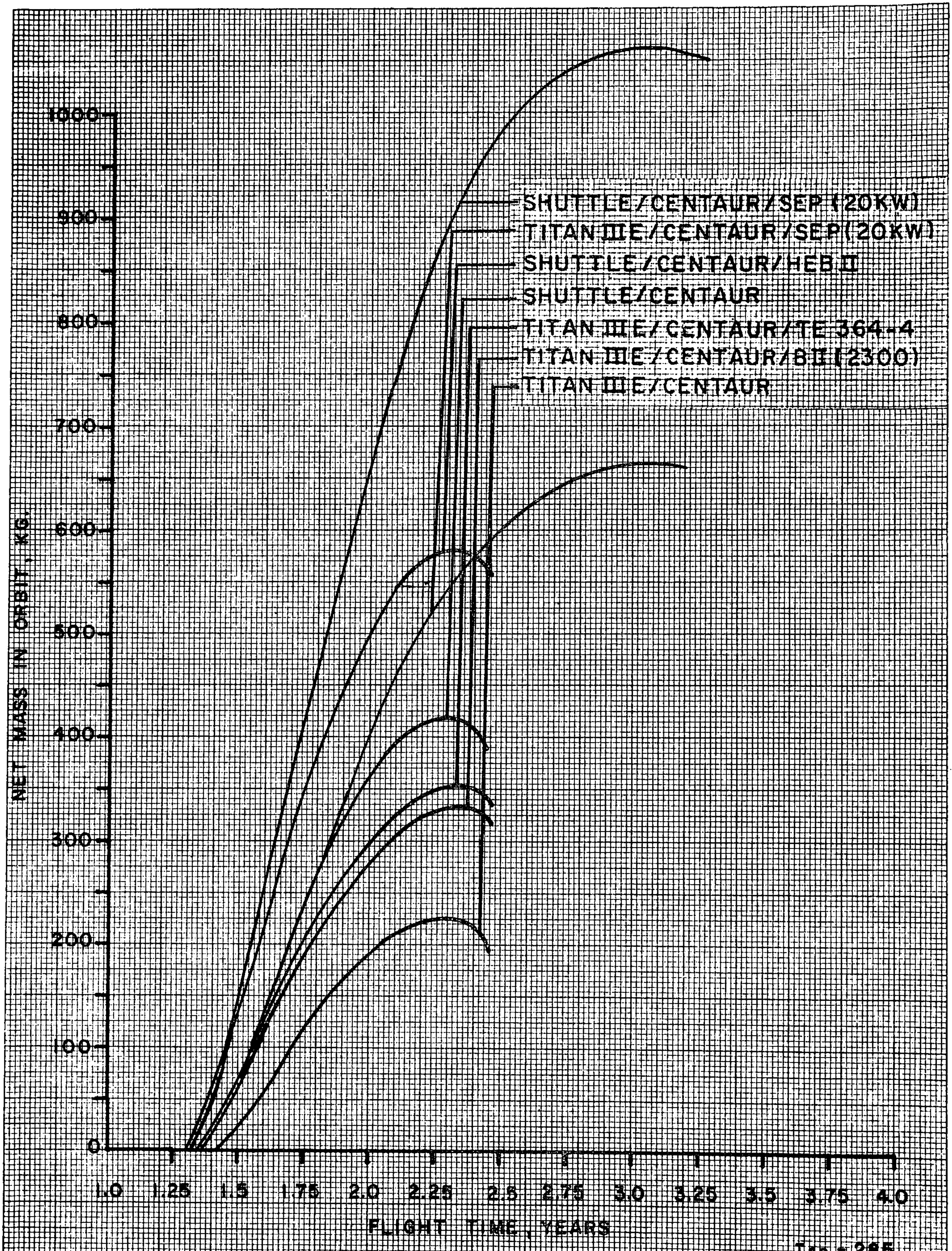


FIGURE 33. 1980 JUPITER 15^d ORBITER

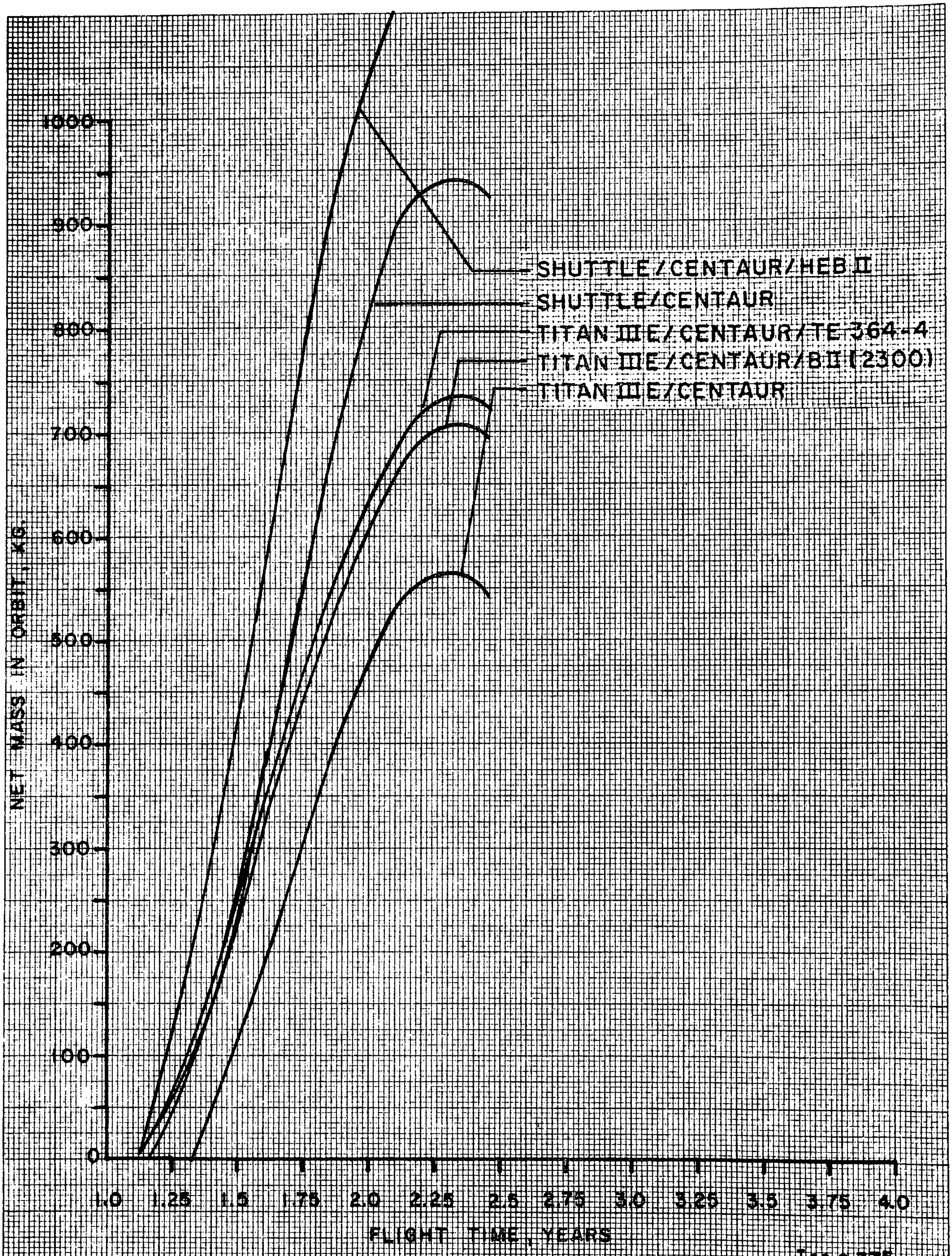


FIGURE 34. 1981-2 JUPITER 60^d ORBITER

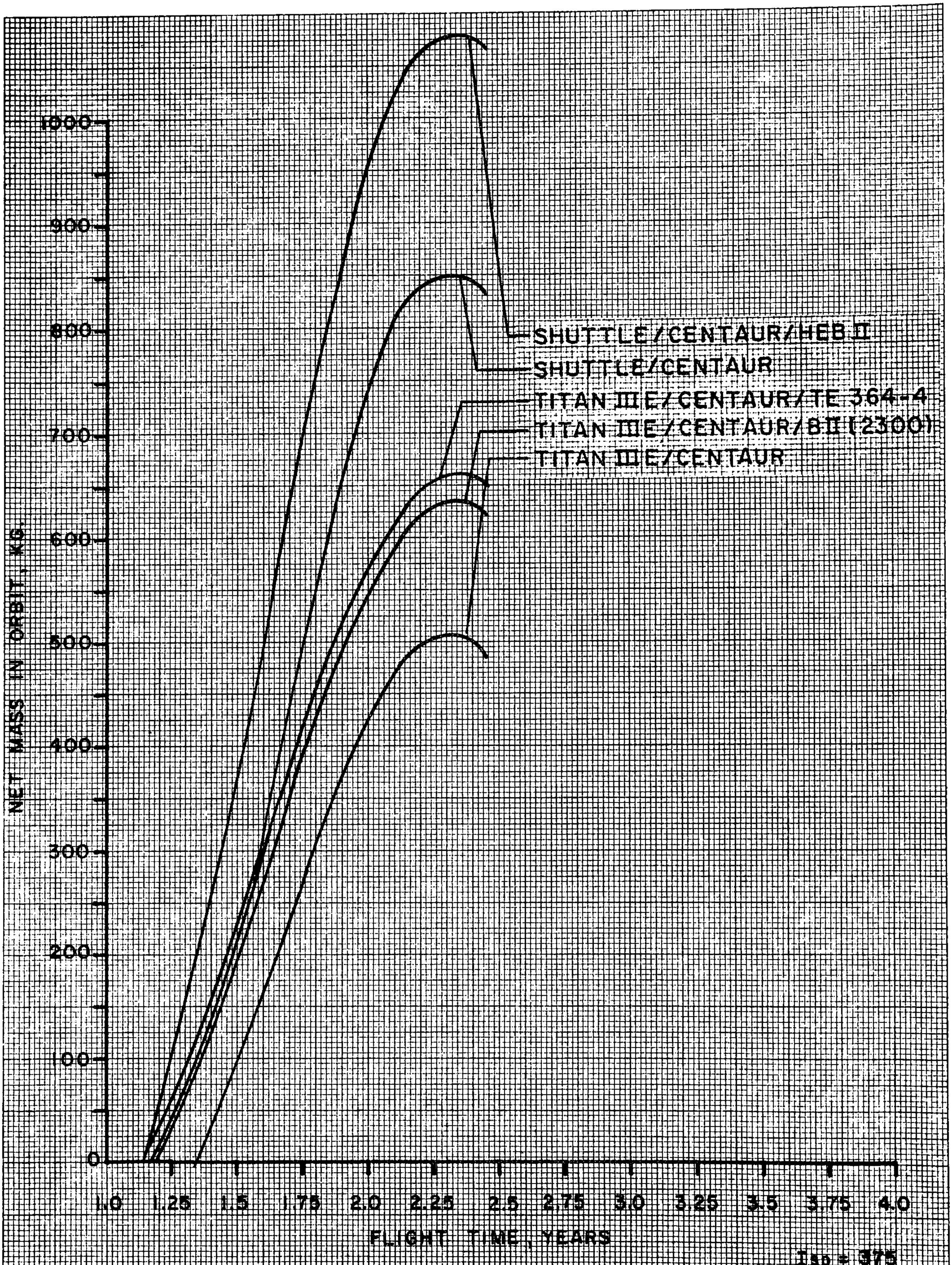


FIGURE 35. 1981-2 JUPITER 30^d ORBITER

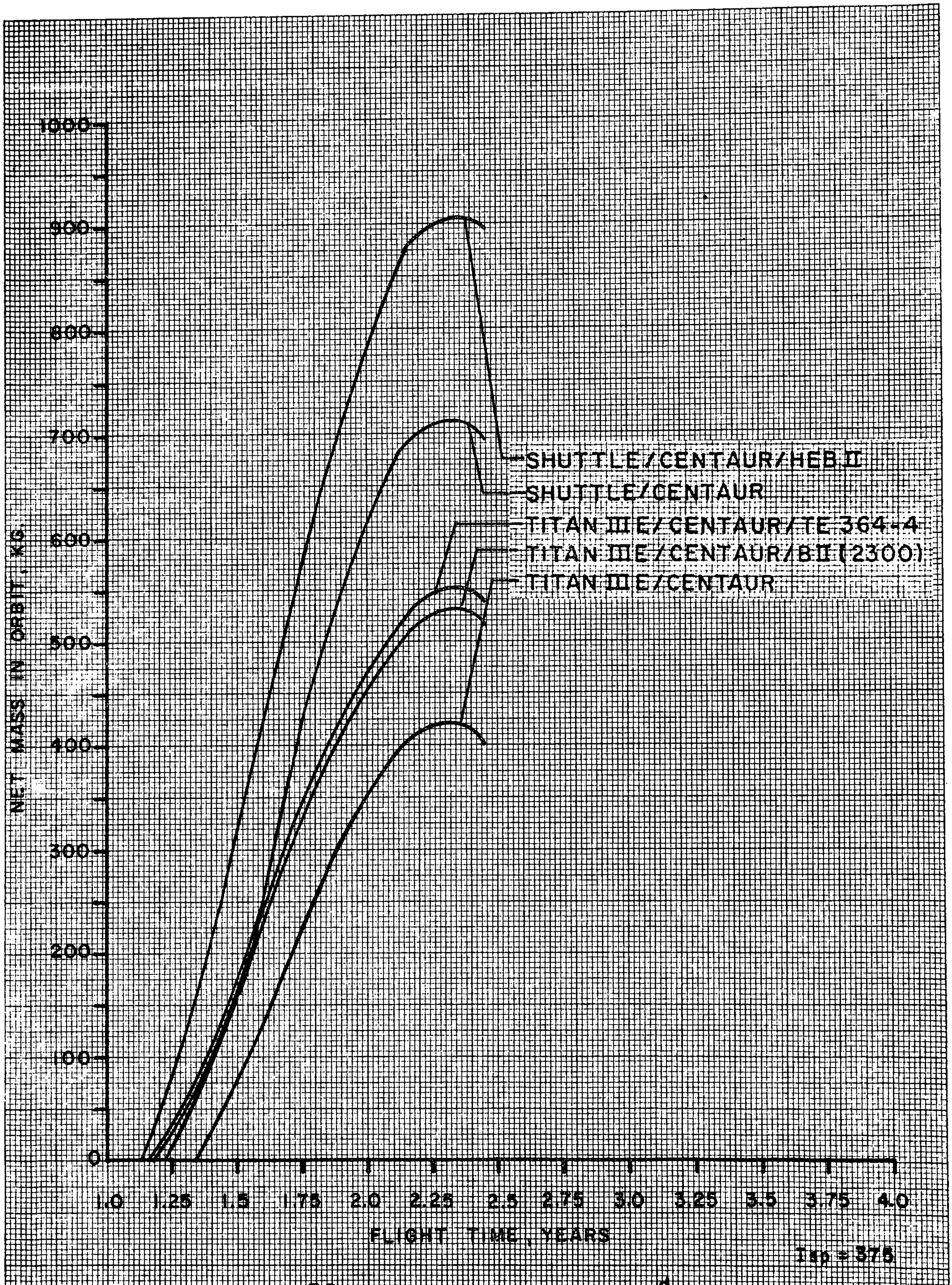


FIGURE 36. 1981-2 JUPITER 15° ORBITER

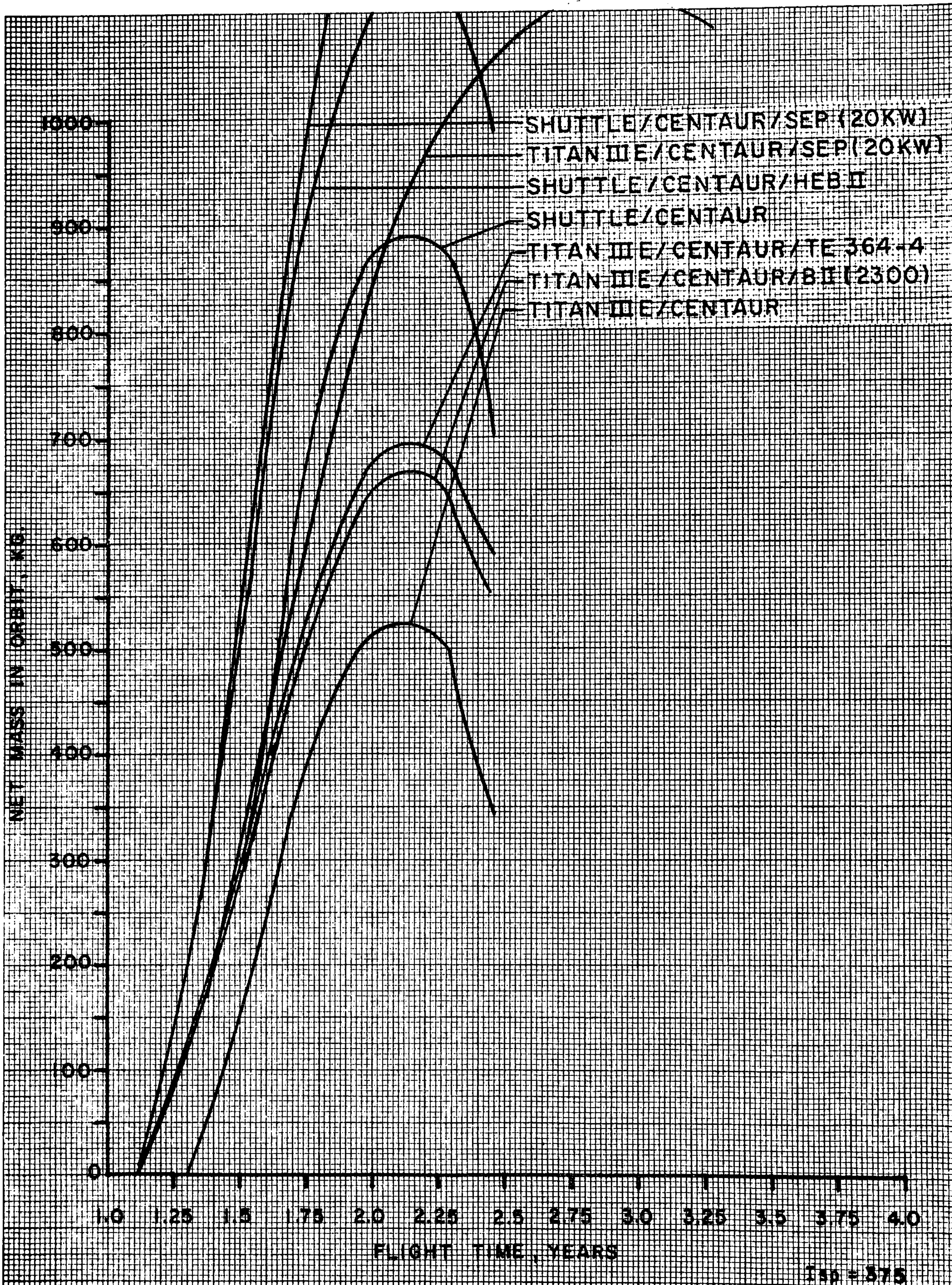


FIGURE 37. 1983 JUPITER 60^d ORBITER

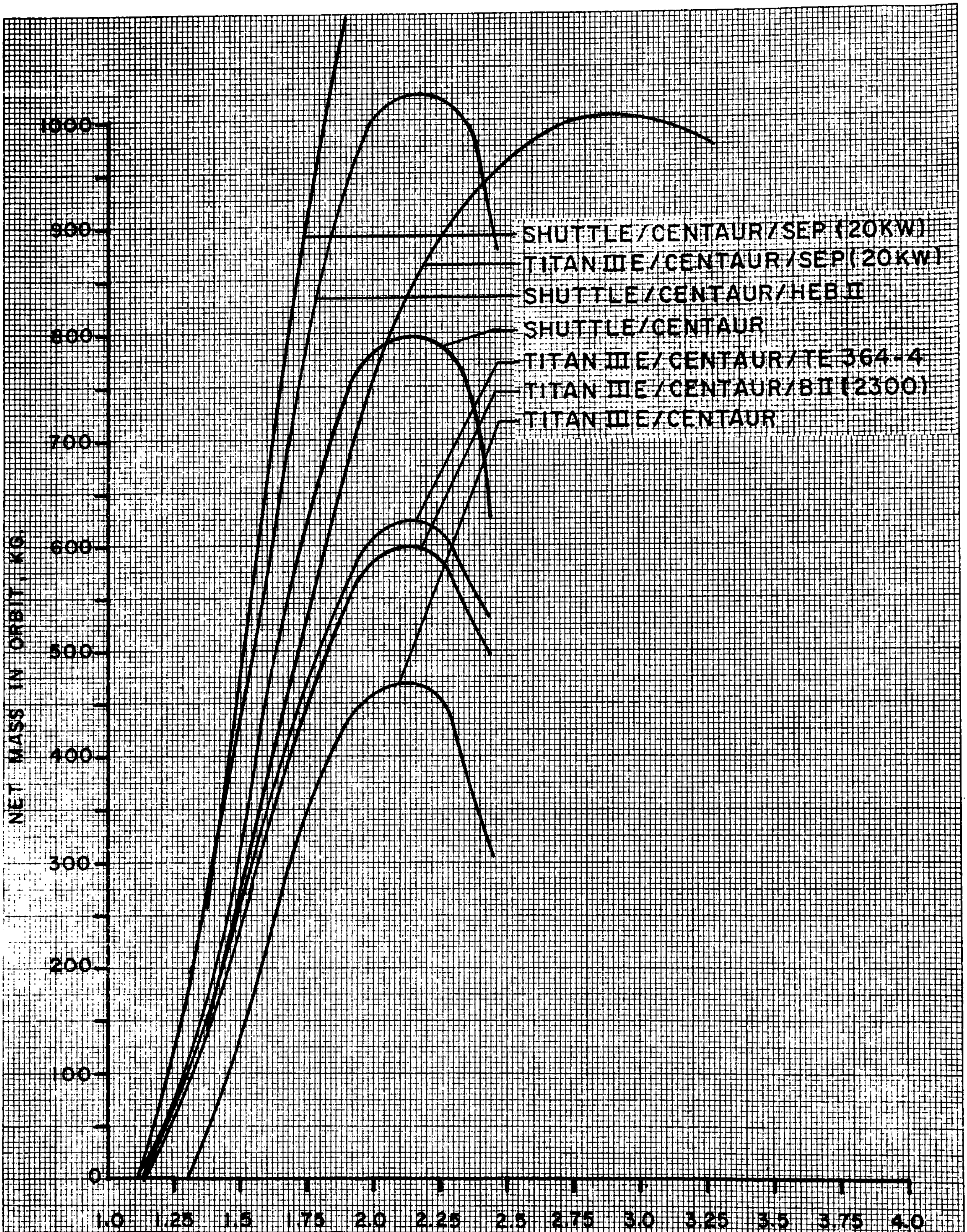


FIGURE 38. 1983 JUPITER 30^d ORBITER

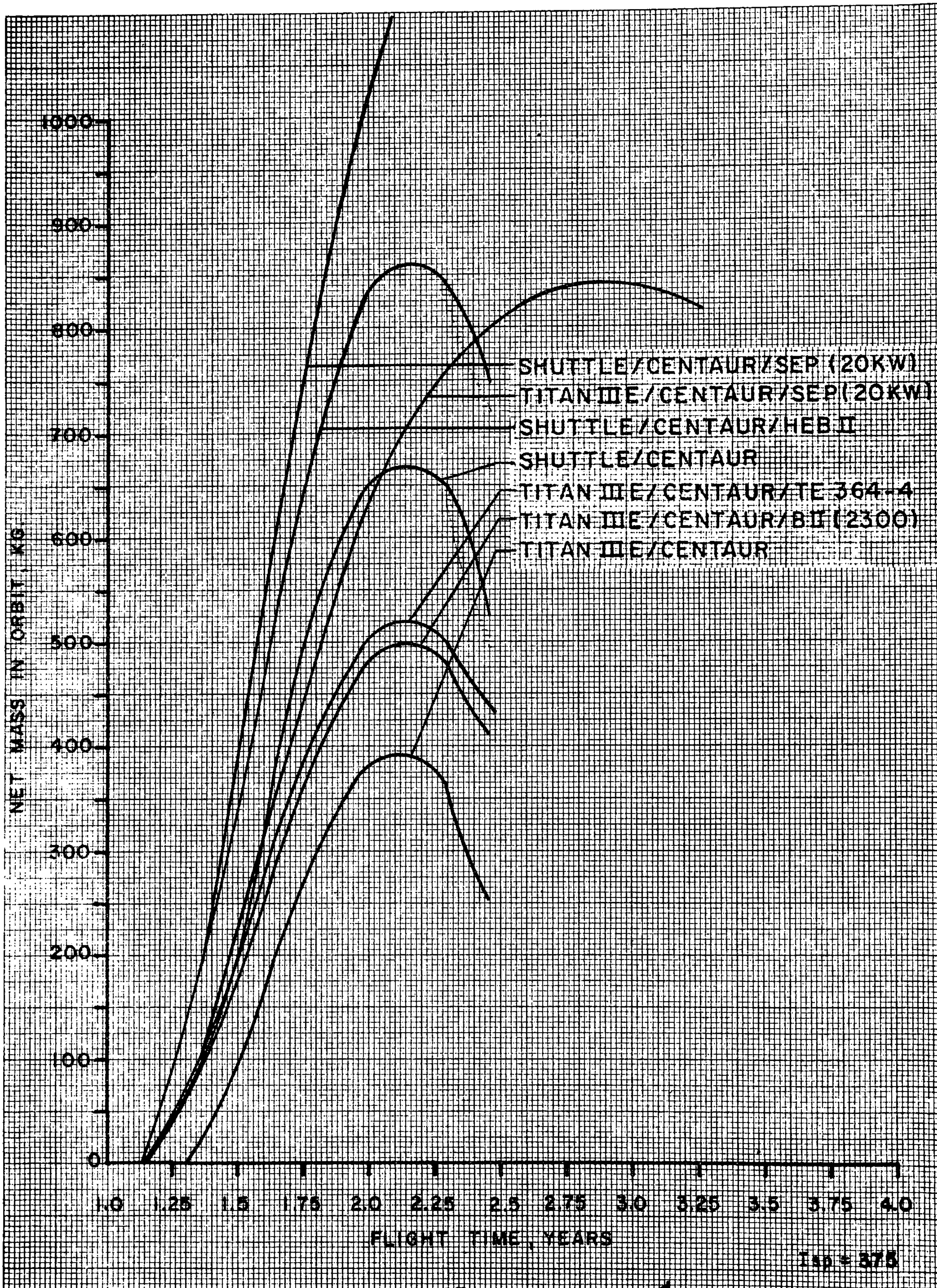


FIGURE 39. 1983 JUPITER 15^d ORBITER

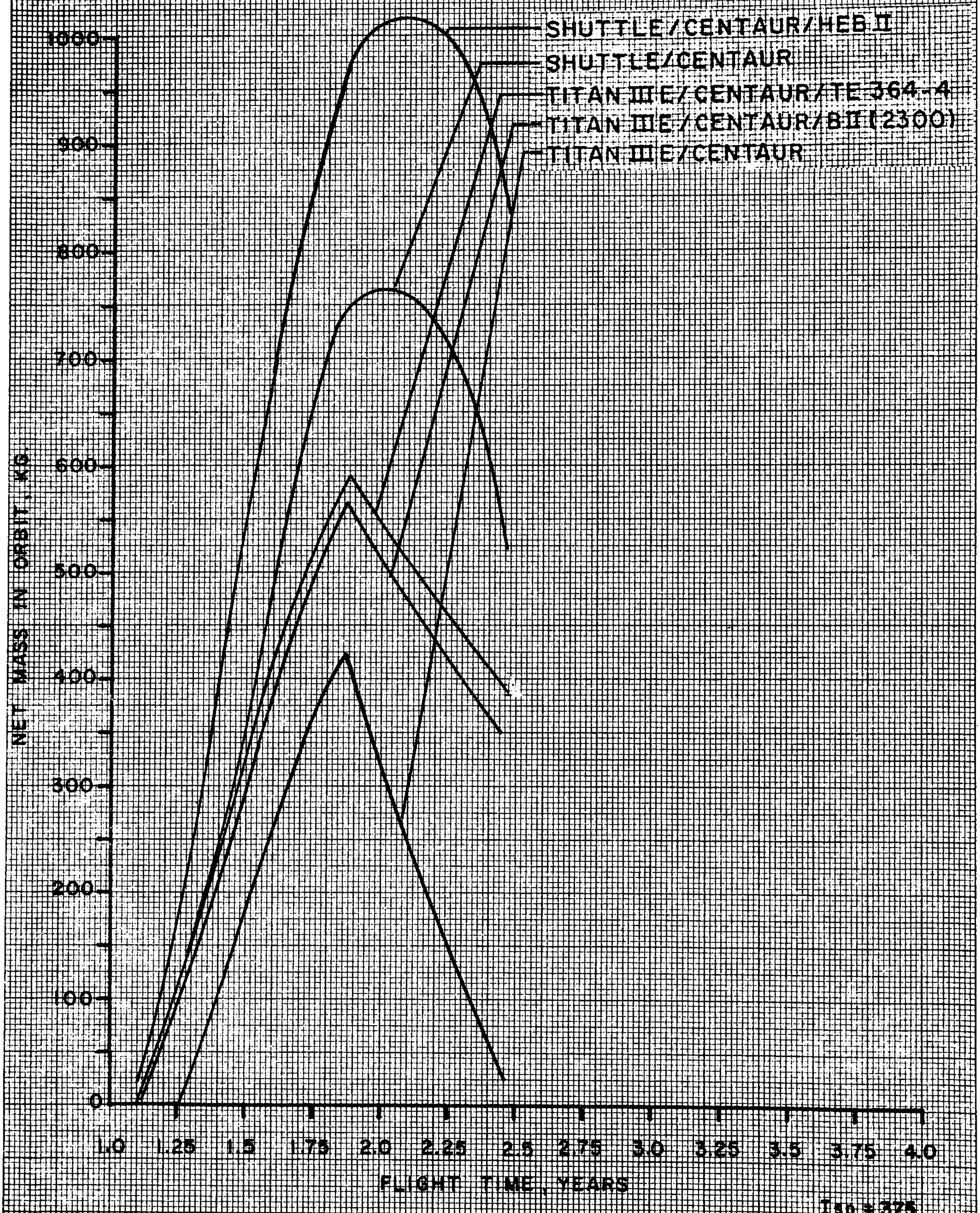


FIGURE 40. 1984 JUPITER 60° ORBITER

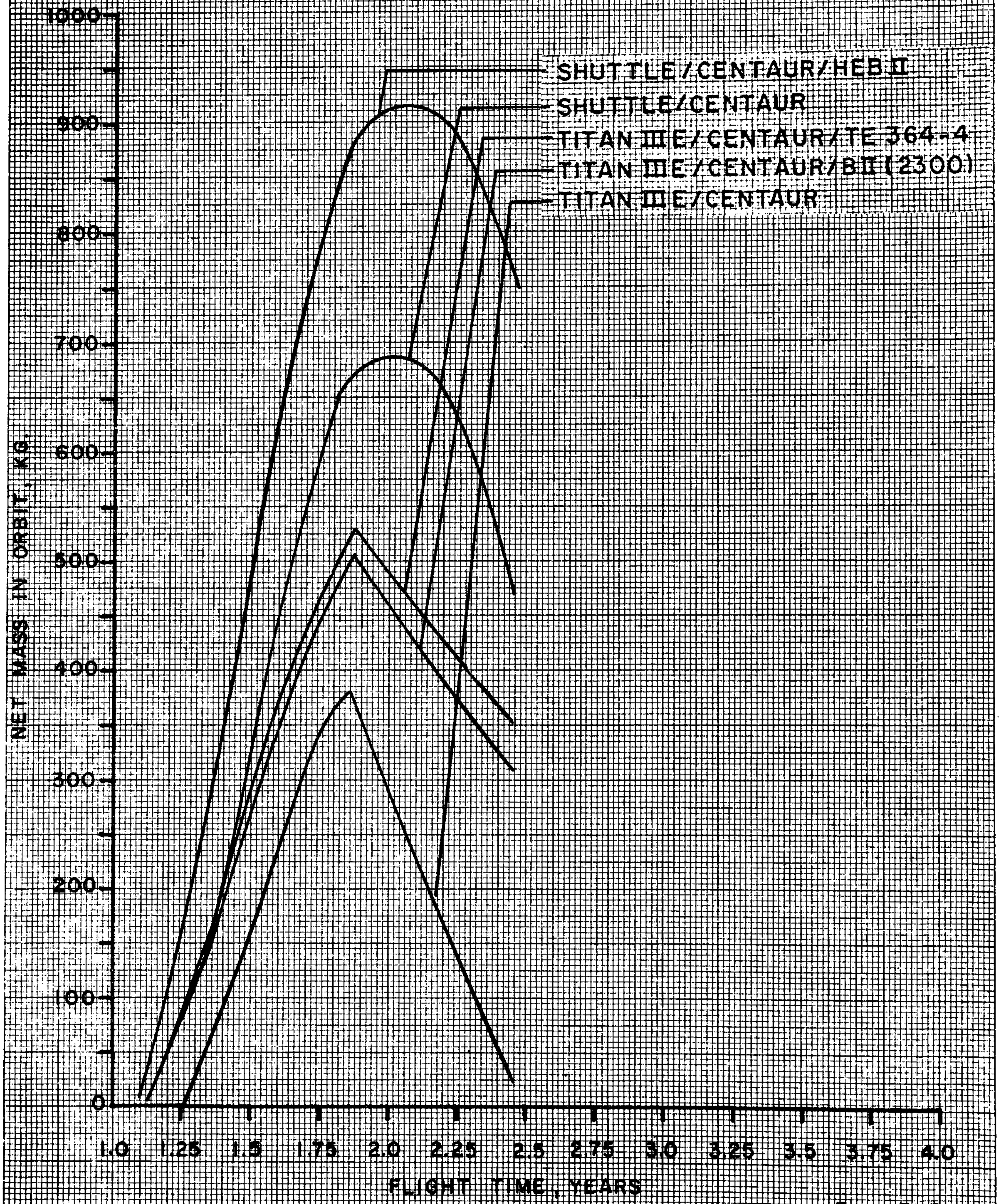


FIGURE 41. 1984 JUPITER 30^d ORBITER

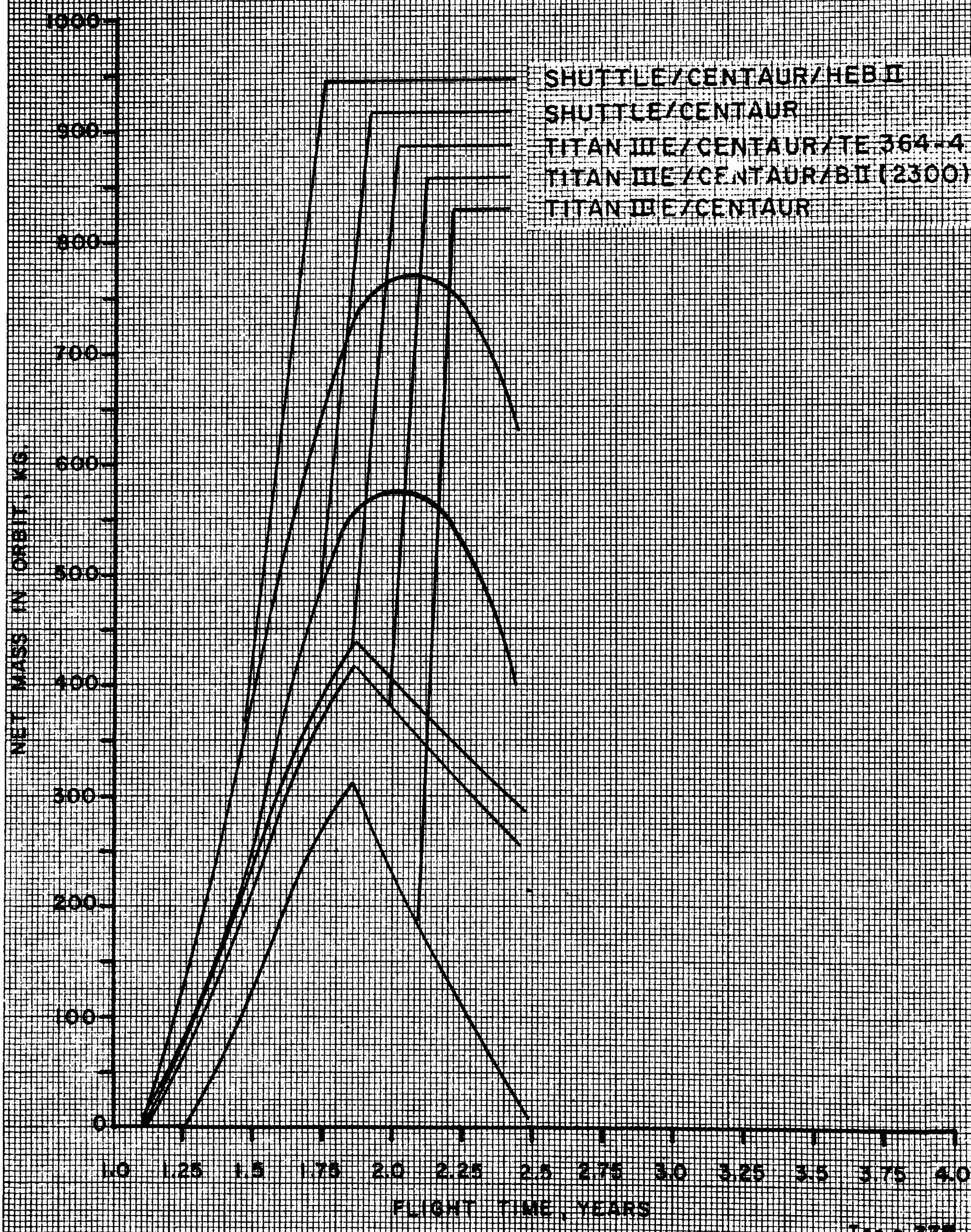


FIGURE 42. 1984 JUPITER 15^{d} ORBITER

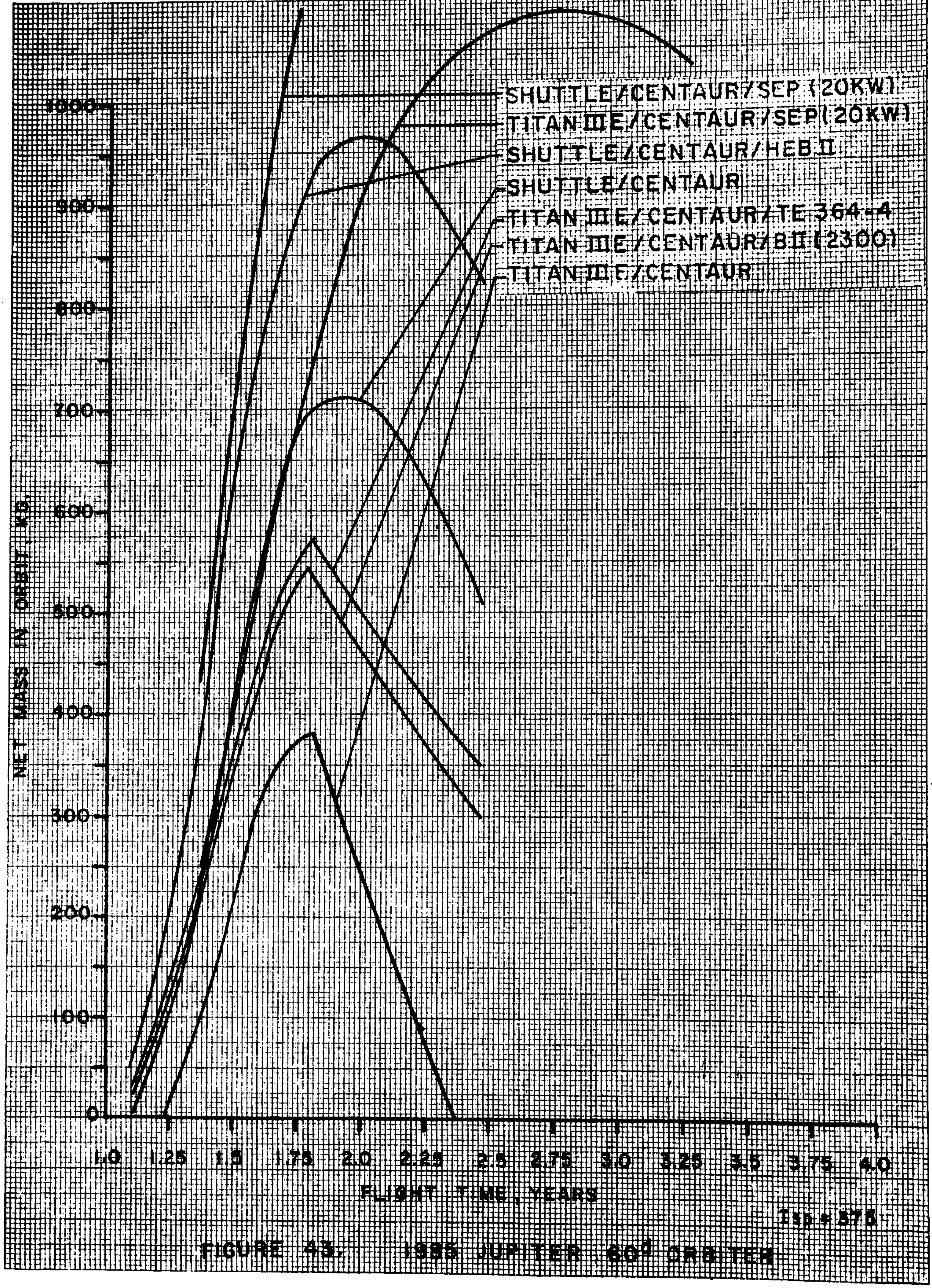


FIGURE 43. 1985 JUPITER 60° ORBITER

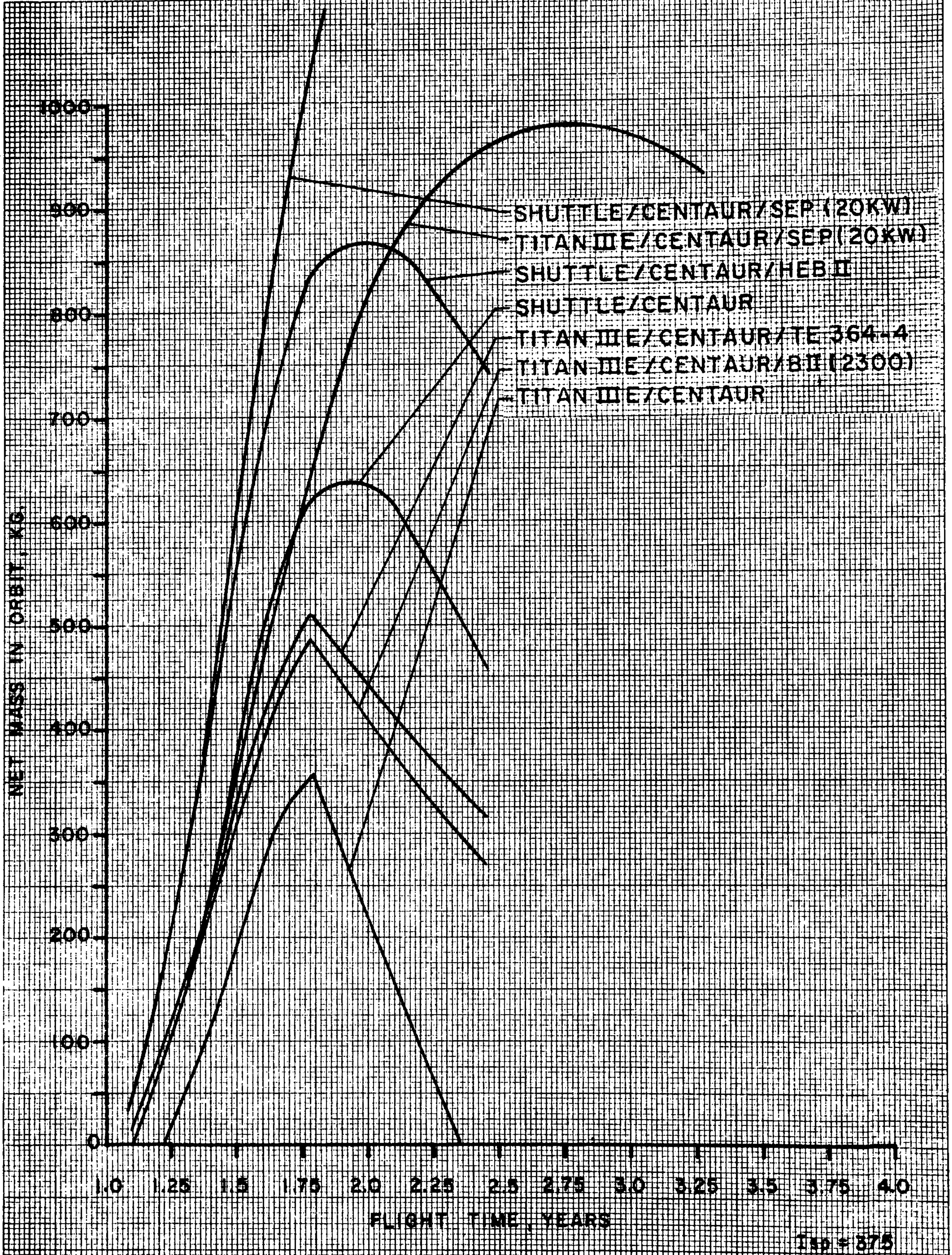


FIGURE 44. 1985 JUPITER 30^d ORBITER

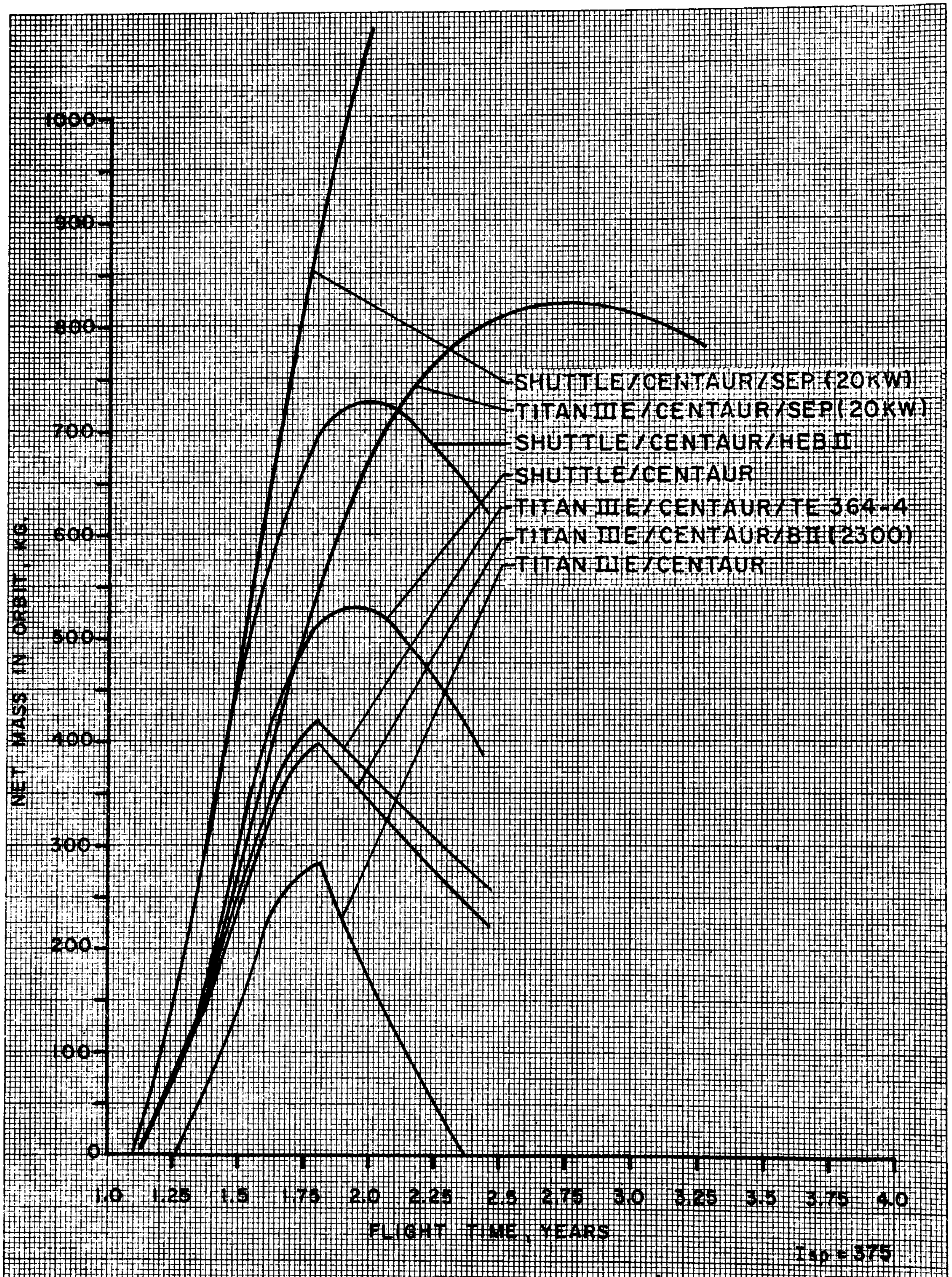


FIGURE 45. 1985 JUPITER 15^{d} ORBITER

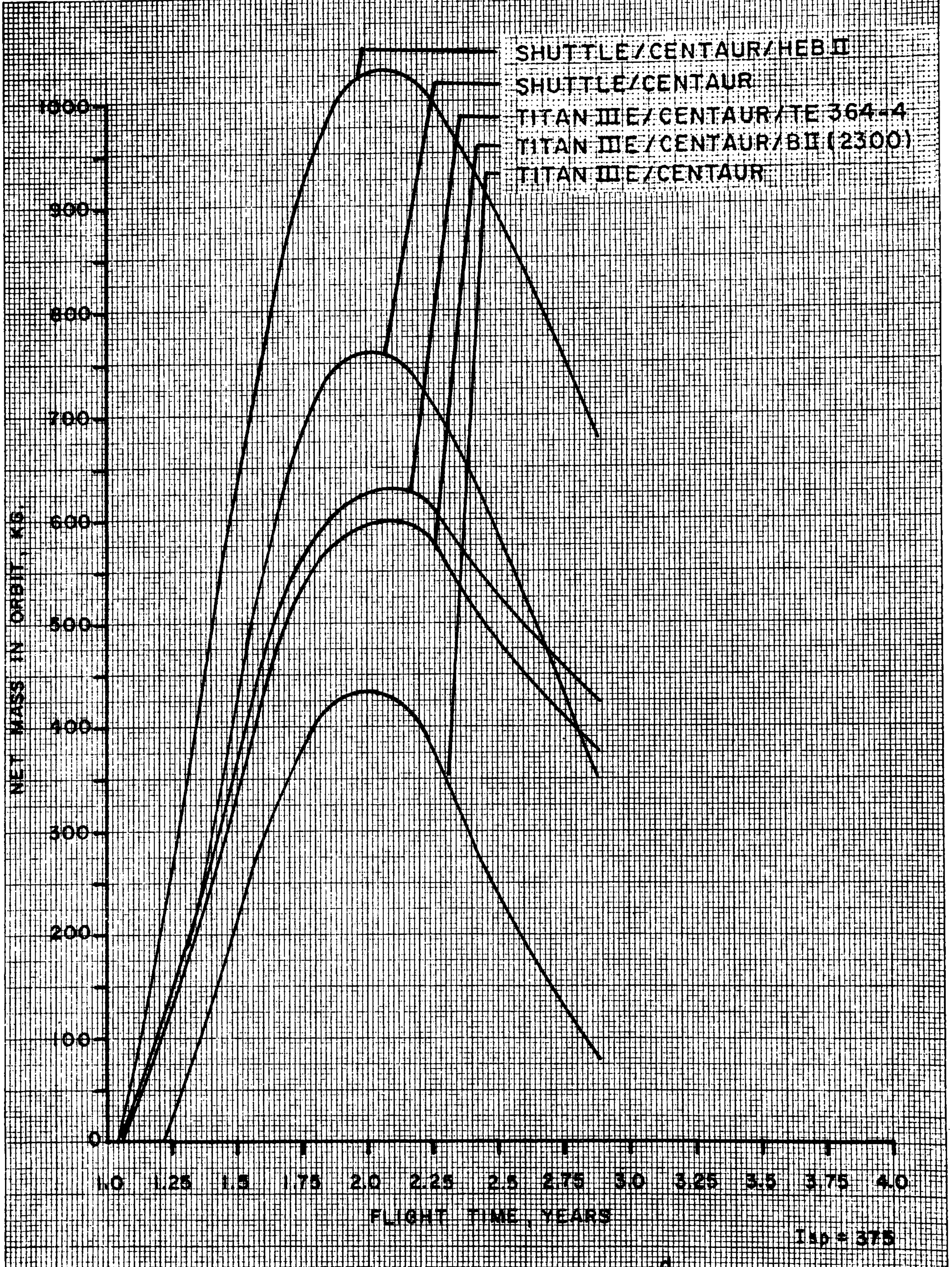


FIGURE 46. 1986 JUPITER 60° ORBITER

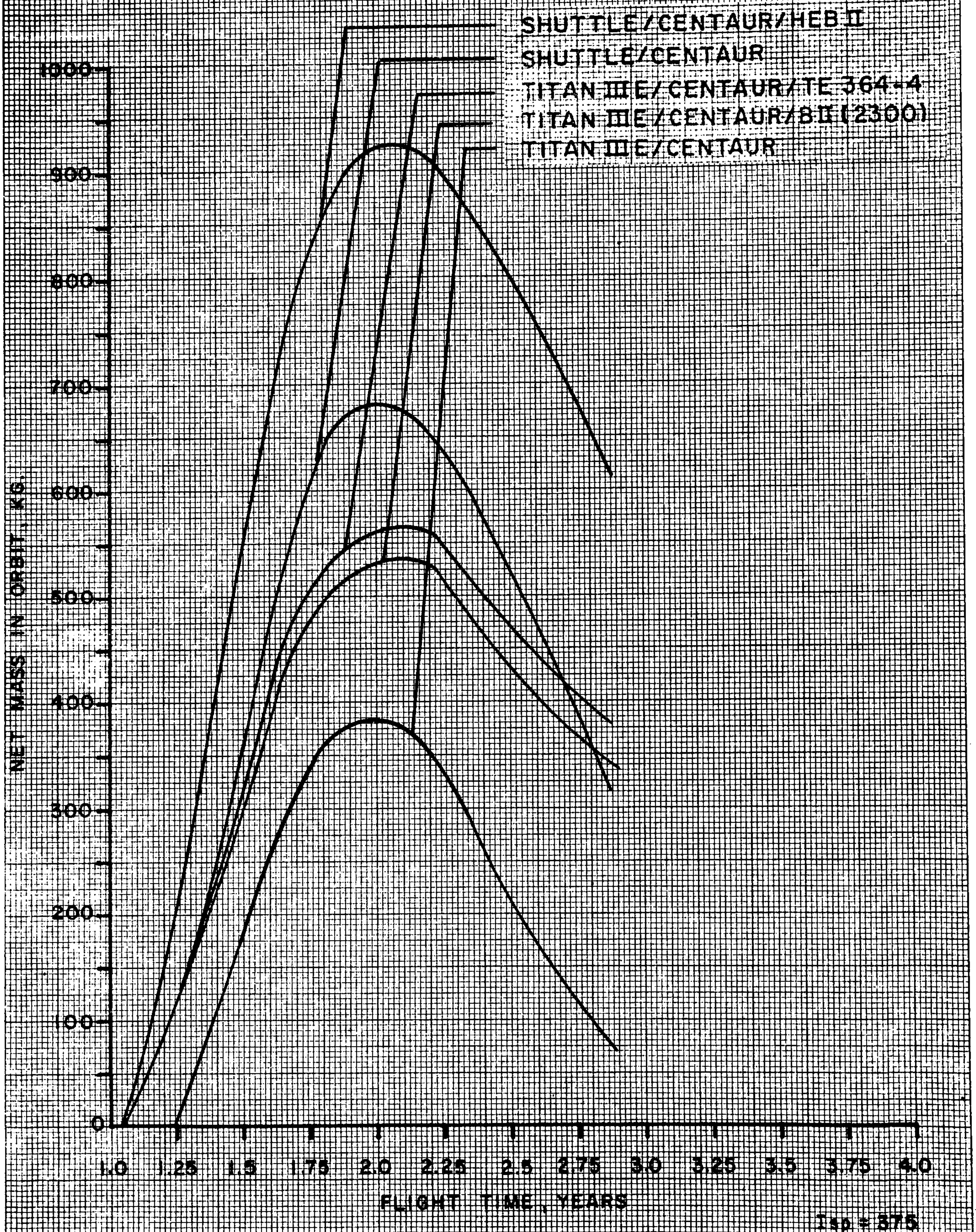


FIGURE 47. 1986 JUPITER 30° ORBITER

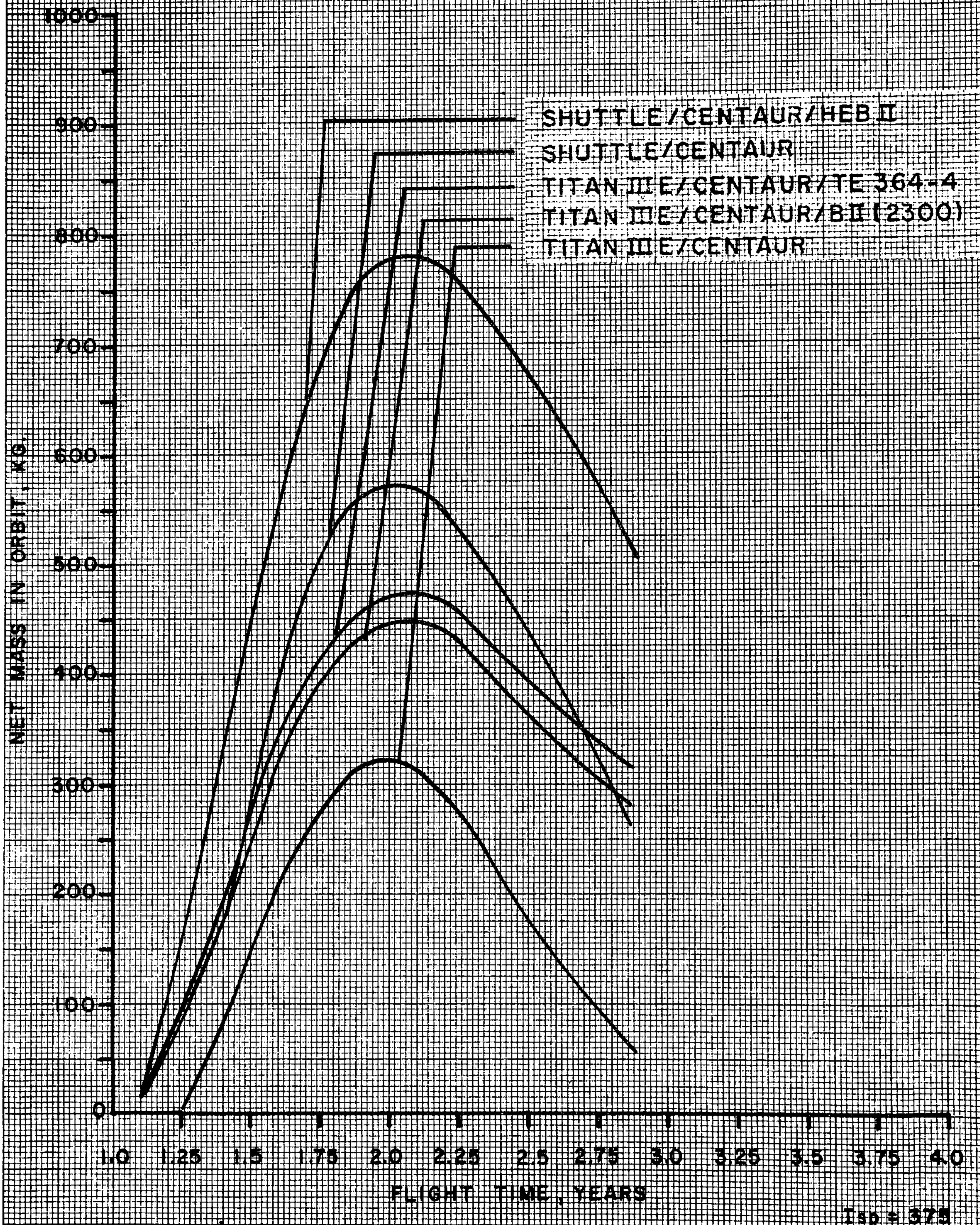


FIGURE 48. 1986 JUPITER 15° ORBITER

3.3 Saturn Flyby Missions: 1976 - 1980

The launch opportunities studied cover about one-third of an Earth-Saturn cycle since Saturn has a 29.5 year period. This interval is one of generally decreasing launch velocity requirements. For ballistic flyby missions the net approach mass should equal the spacecraft mass including the midcourse propulsion system. For SEP flybys any provision for a stage (typically 300 kg) should also be added.

Between 1976 and 1979 the maximum payload which can be launched direct to Saturn using a Titan III E/Centaur/TE364-4 vehicle is 450 kg. The DLA constraint causes the sharp peak in these performance curves and reduces the flyby payload by about 50 kg in 1976 and 1977. During the 1980's as the maximum performance shifts to longer flight times (about 4.5 years), the net approach mass for this Titan vehicle comes close to 750 kg (in 1985 and 1986).

The Shuttle typically places 750 kg on a 2.9 to 3.2 year flyby trajectory to Saturn in the 1980's. Maximum performance is about 1100 kg using longer flight times. Solar electric payload results are less sensitive to launch opportunity. Using 300 kg for the SEP stage weight, a net approach mass of 750 kg is obtained with the Titan launched 20 kw SEP system at 3.2 years and the maximum payload is over 1100 kg. (This is very similar to the Shuttle performance). Using the Shuttle to launch the SEP can reduce the flight time for 750 kg net to 2.6 years or raise the maximum payload to about 1750 kg.

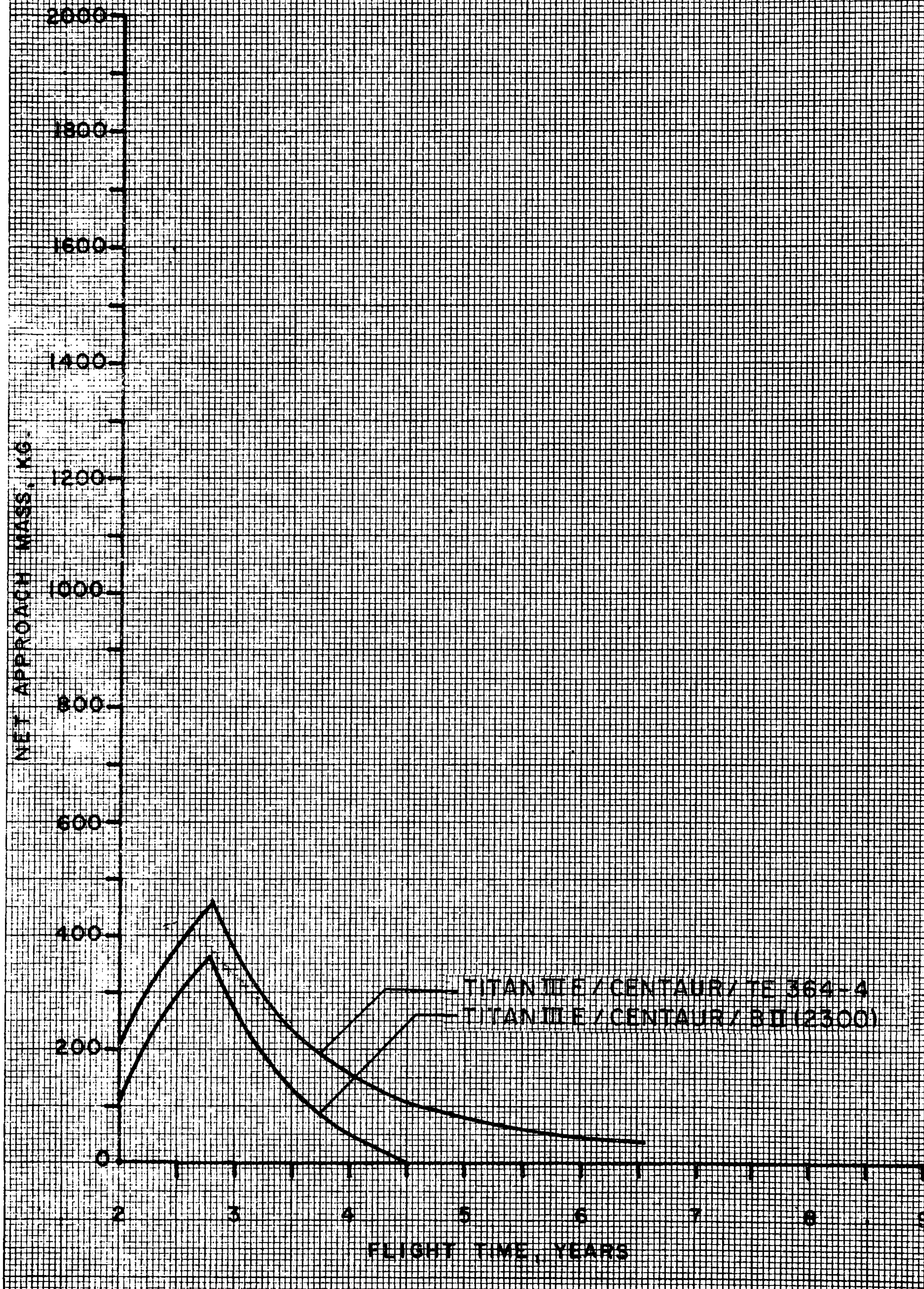


FIGURE 49. 1976 SATURN FLYBY

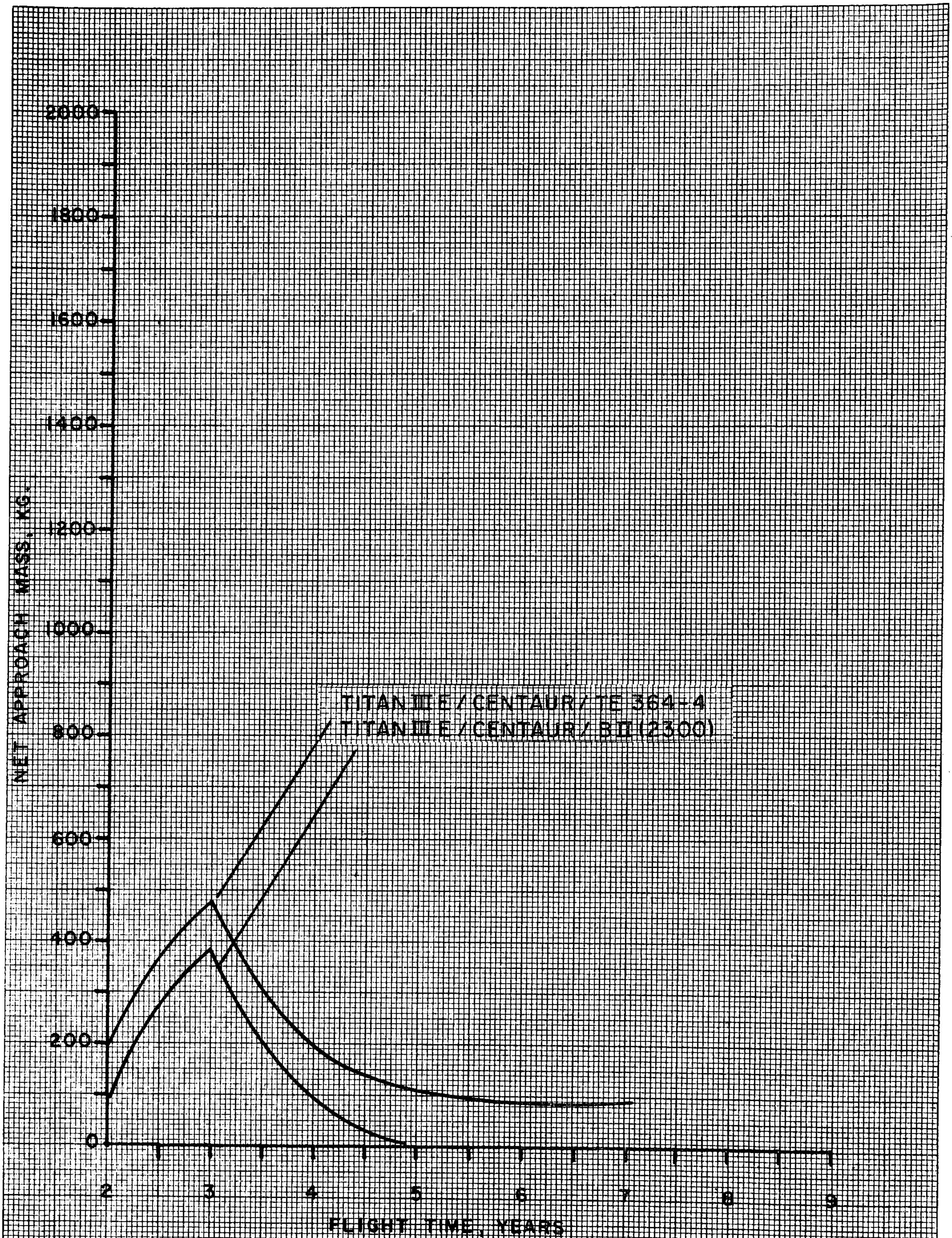


FIGURE 50. 1977 SATURN FLYBY

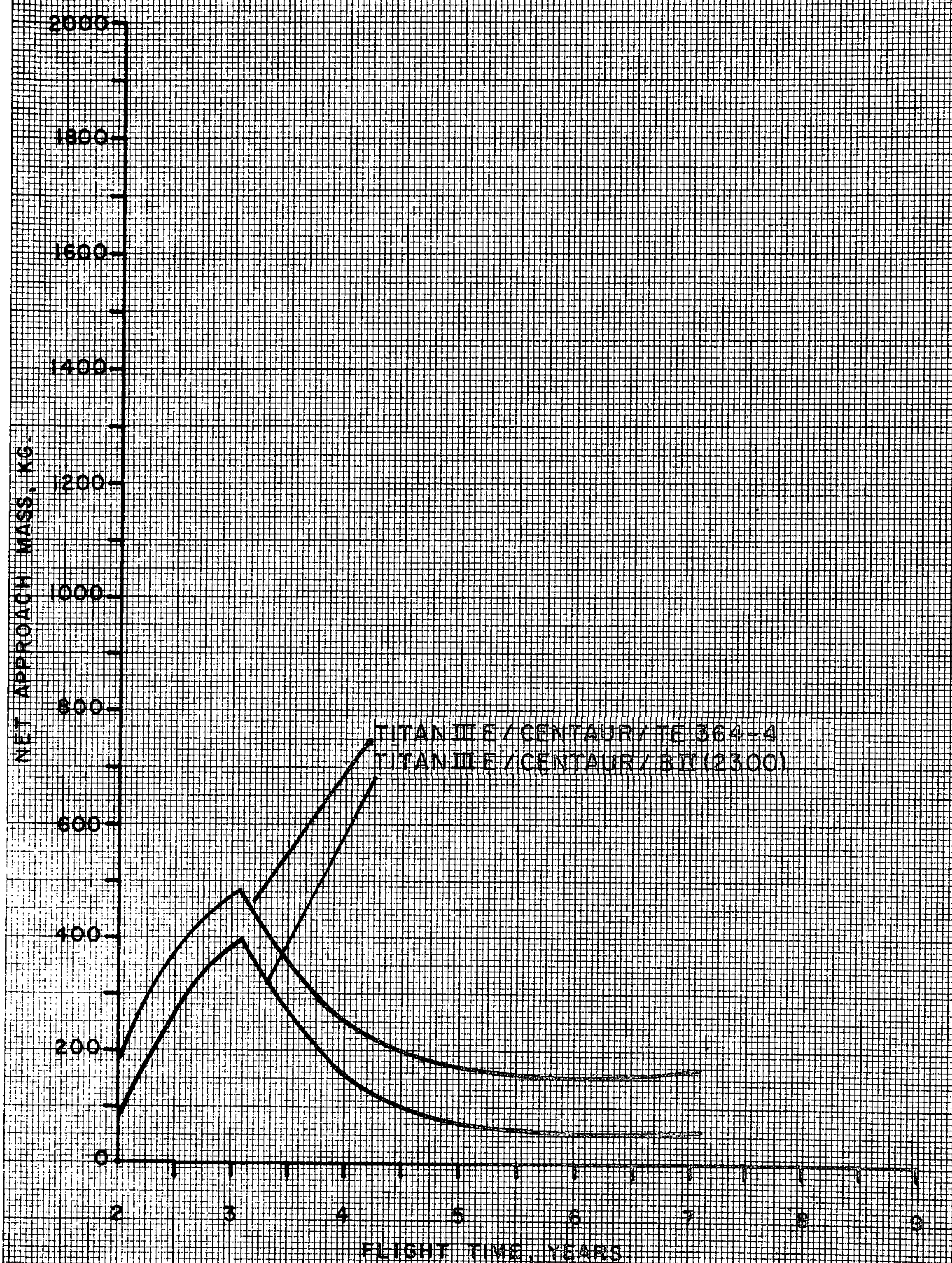


FIGURE 51. 1978 SATURN FLYBY

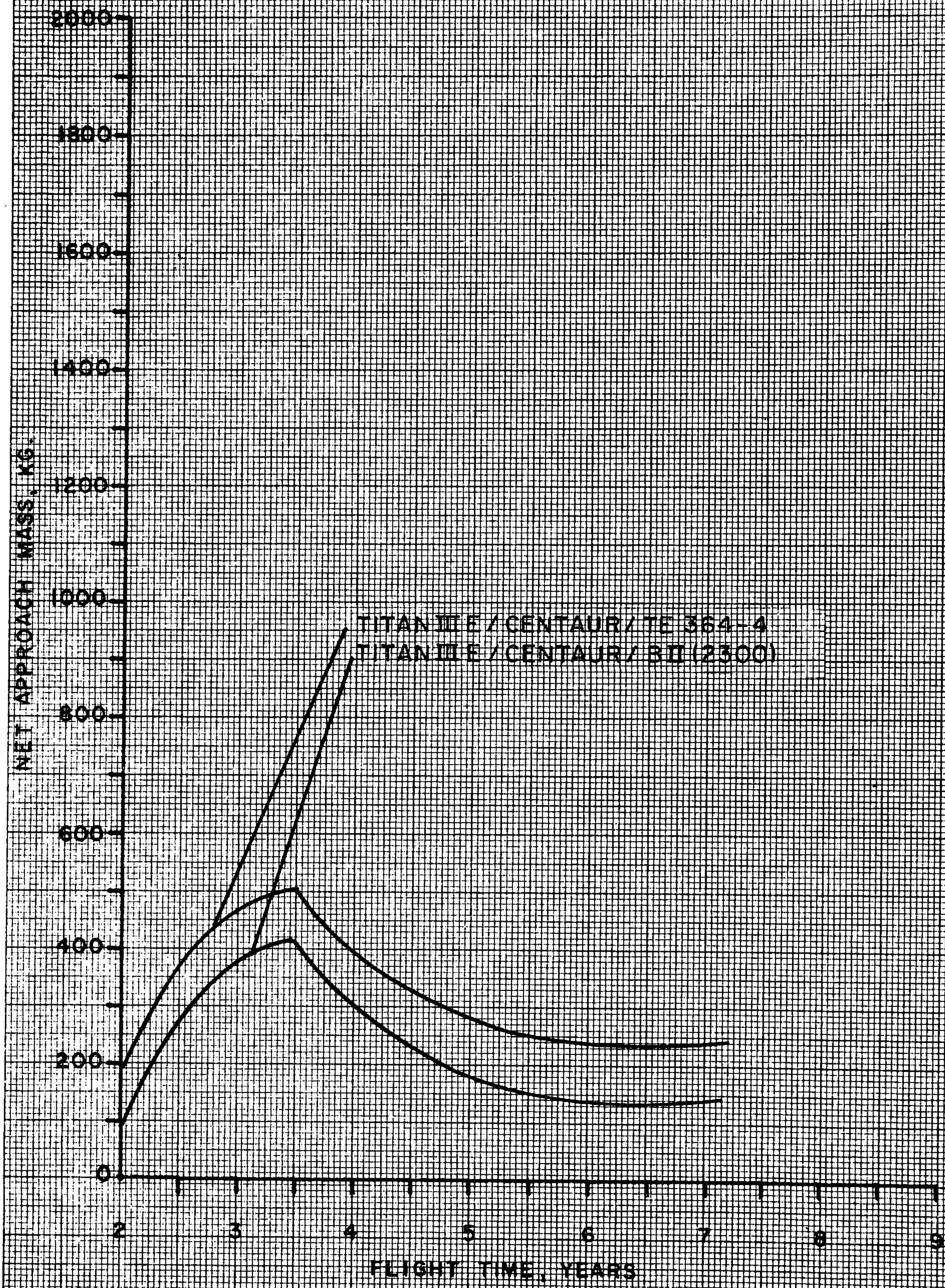


FIGURE 52. 1979 SATURN FLYBY

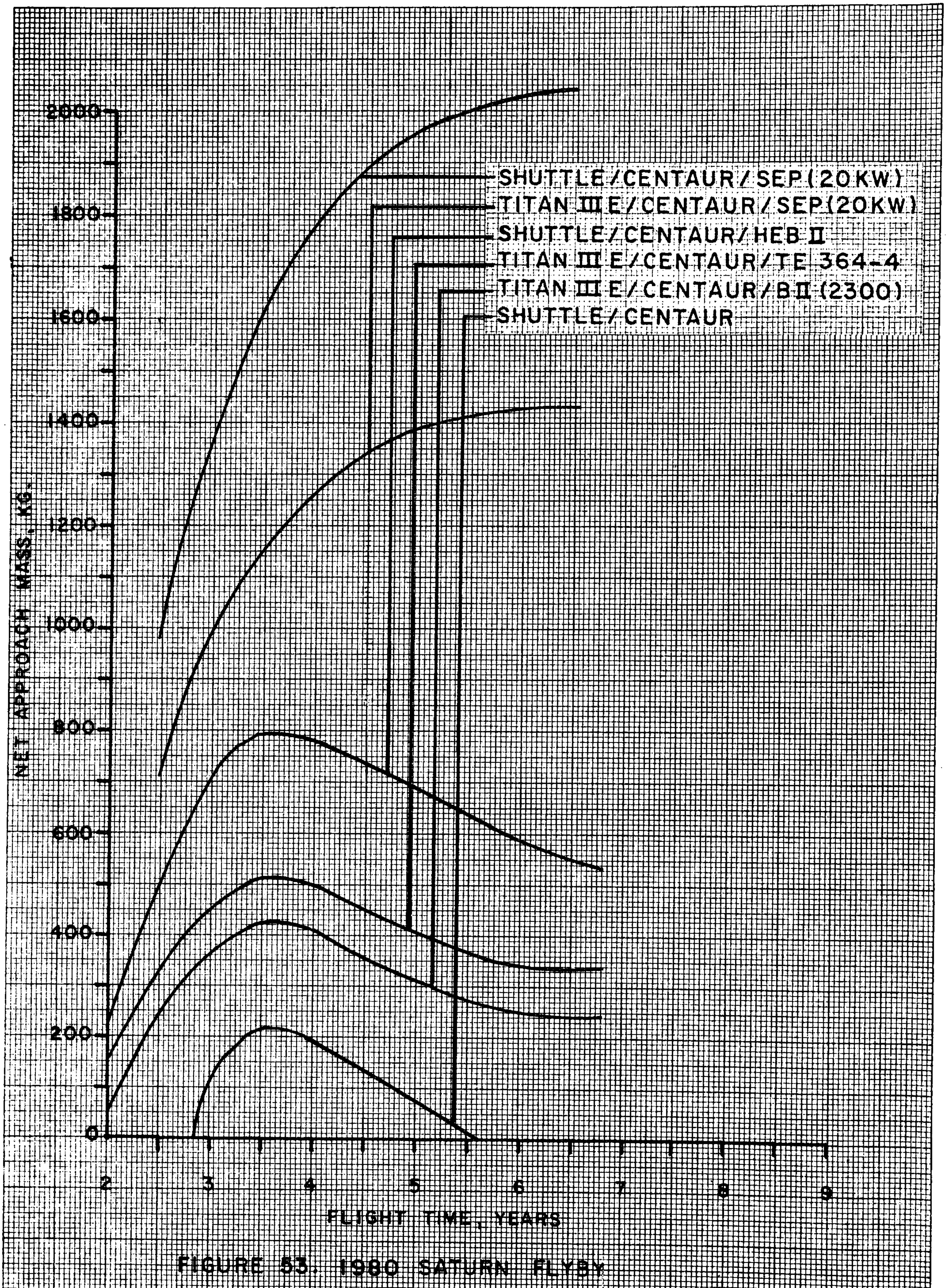


FIGURE 53. 1980 SATURN FLYBY

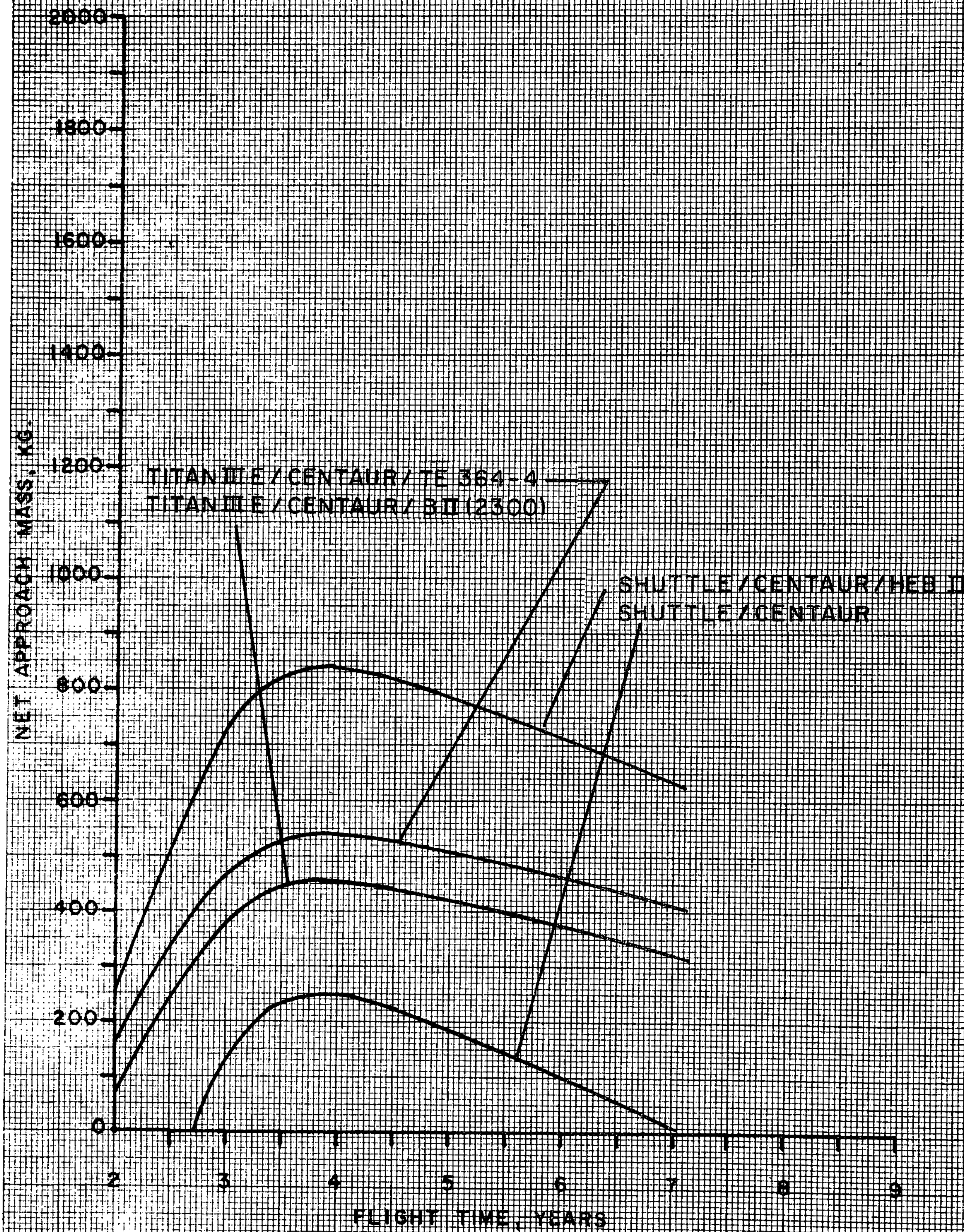


FIGURE 54. 1981 SATURN FLYBY

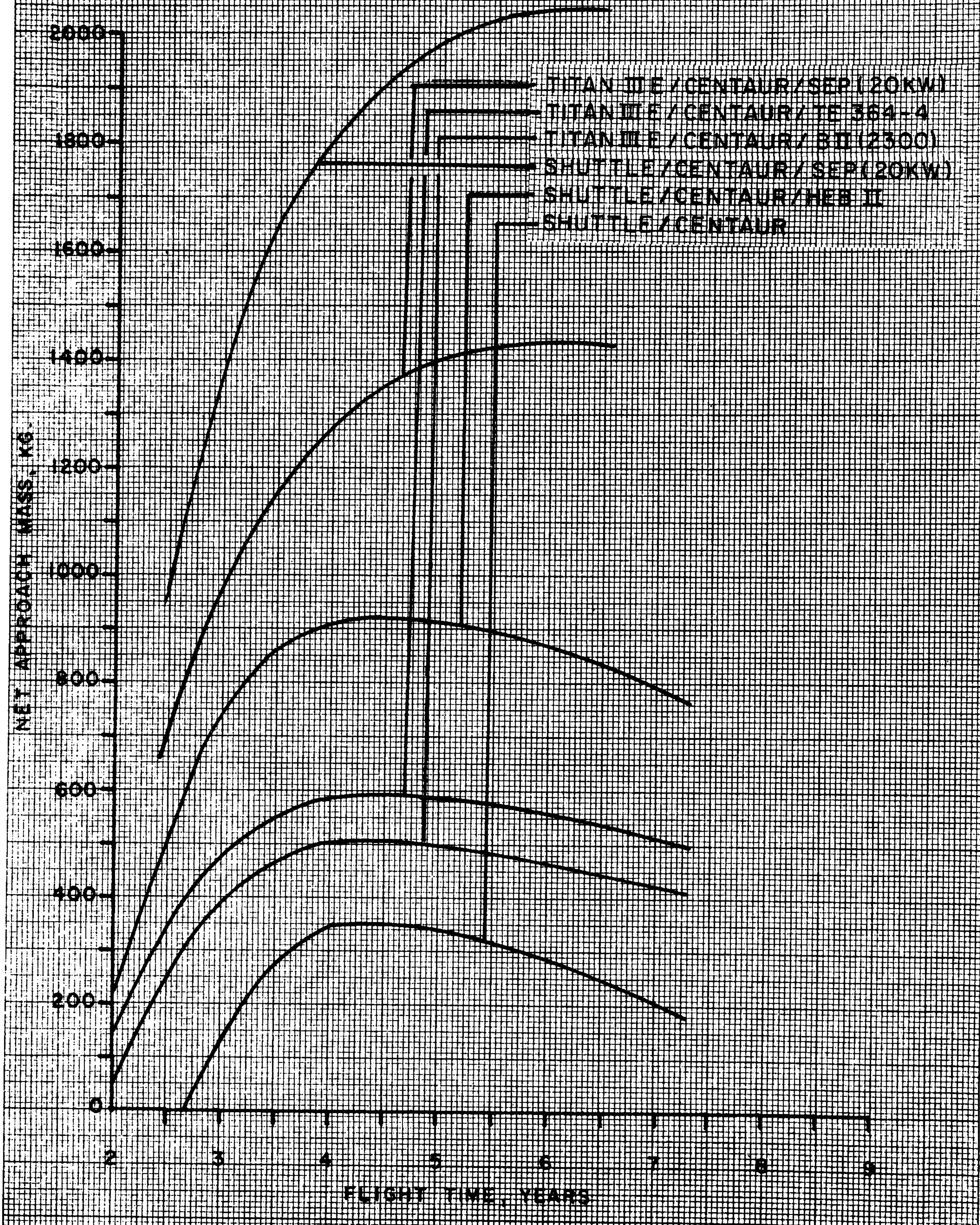


FIGURE 55. 1982 SATURN FLYBY

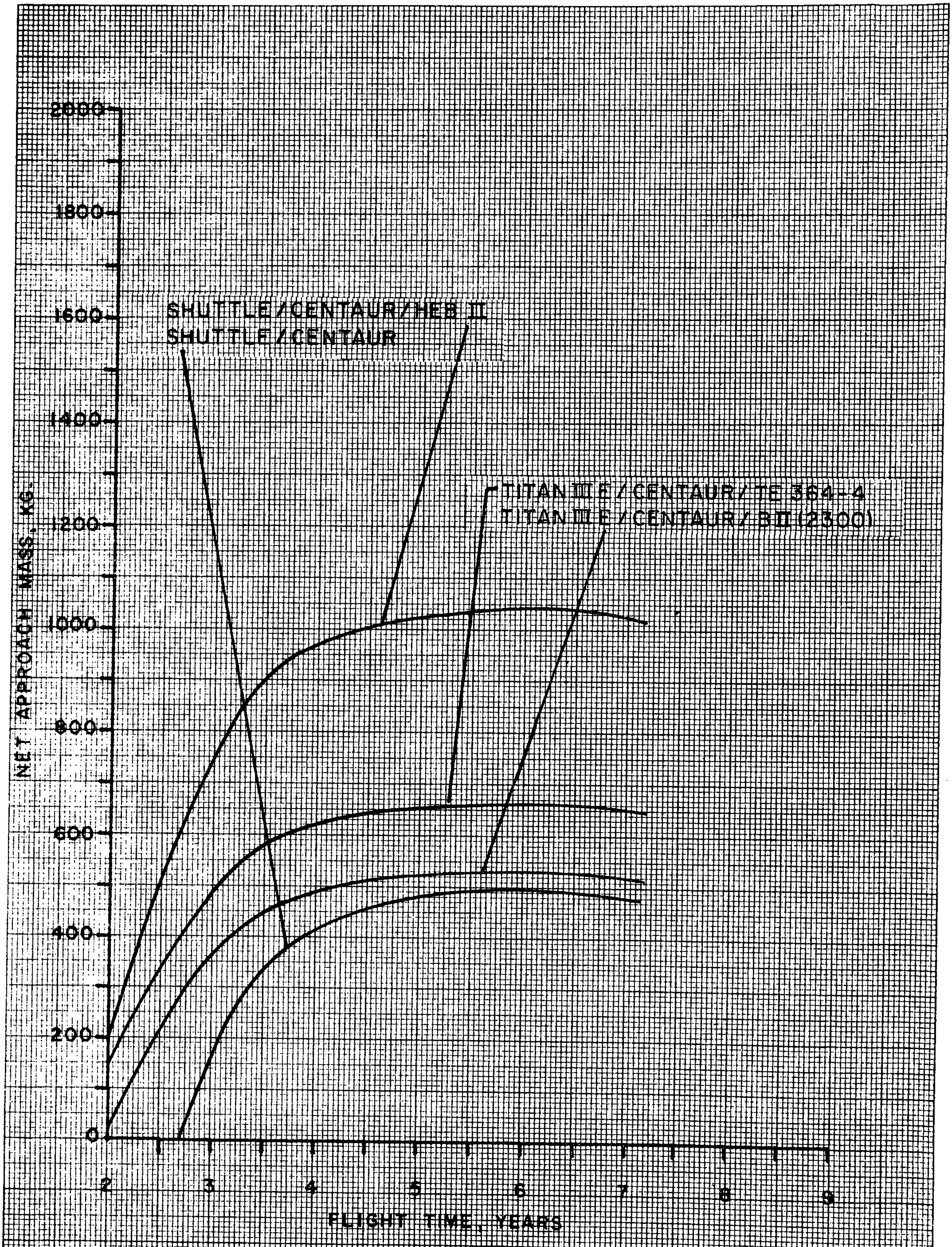


FIGURE 56. 1983 SATURN FLYBY

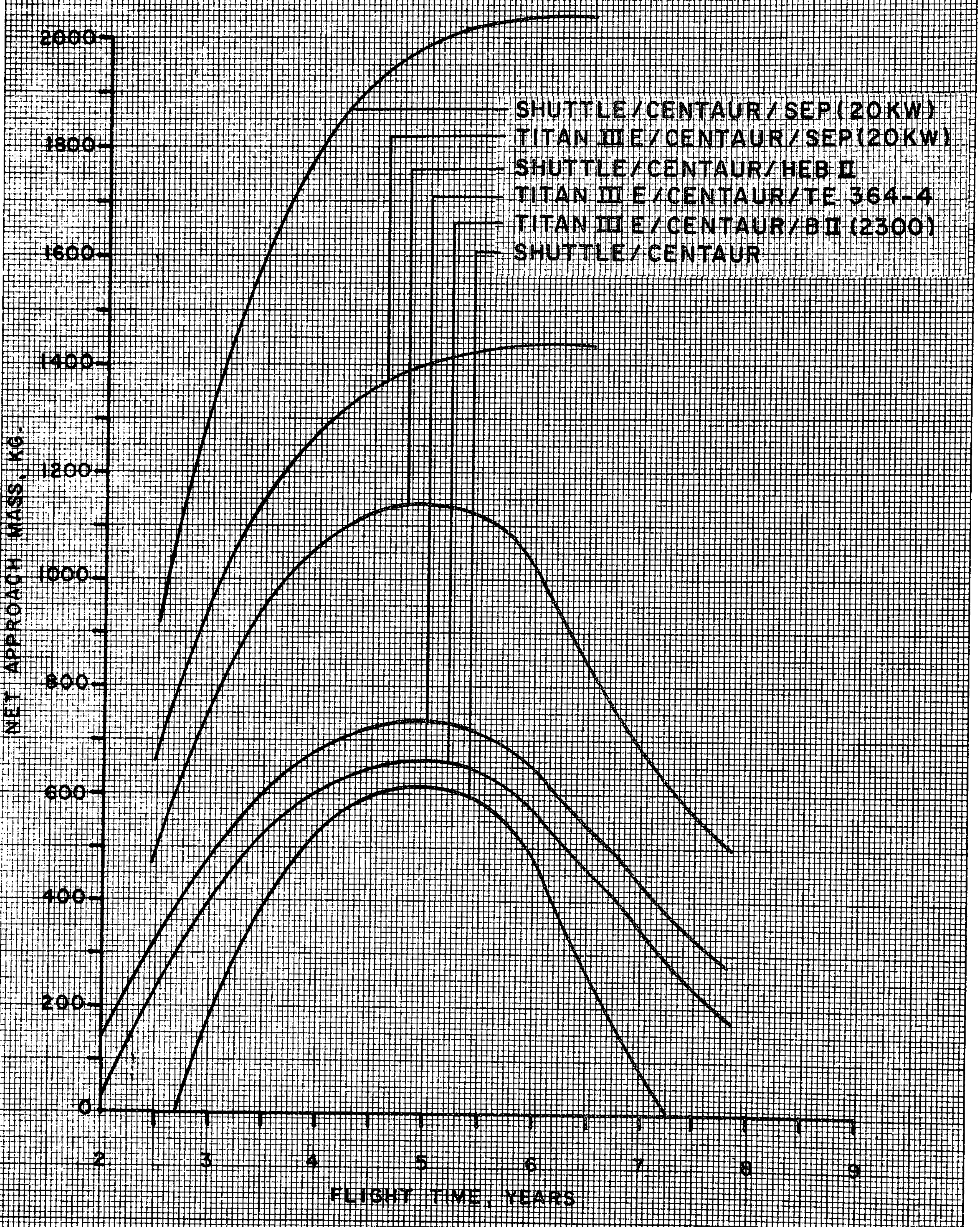


FIGURE 57. 1985 SATURN FLYBY

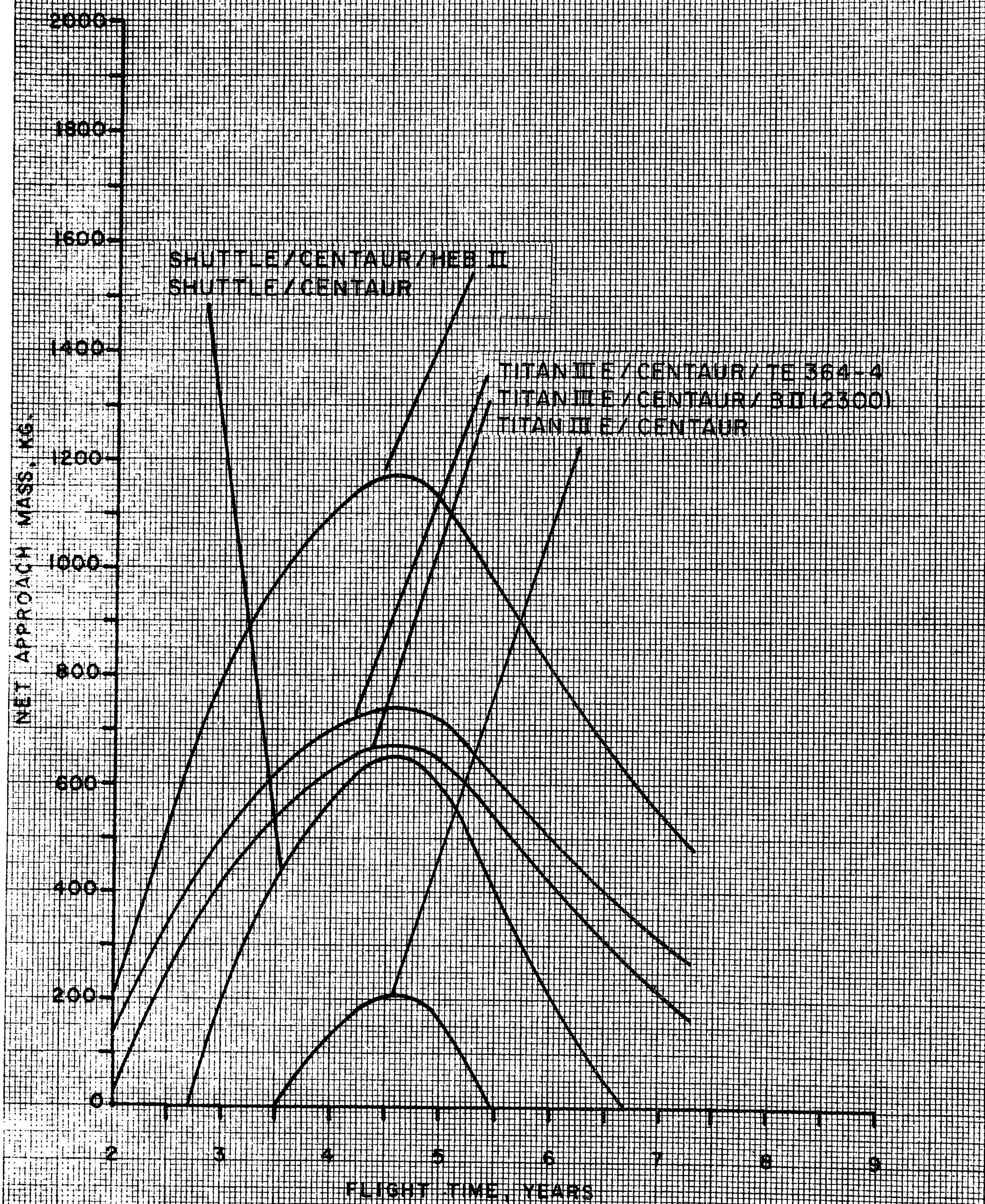


FIGURE 58. 1986 SATURN FLYBY

3.4 Saturn Orbiter Missions: 1980 - 1986

During the late 1970's, all the net mass in orbit calculations yielded payloads of less than 200 kg even if the DLA constraint is removed. Thus no graphs are presented for these orbiter missions. Net mass in orbit curves are presented for Saturn missions in the 1980's. These results allocate 250 m/sec for trajectory and orbit corrections and 300 kg for jettisoned SEP stage subsystems.

The orbits considered have a periapse of 3 Saturn radii and periods of 60, 30 and 15 days. Ballistic missions employing a Titan launch vehicle and a space-storable retro propulsion system cannot place more than 450 kg into these orbits. Adding a 20 kw SEP stage to the Titan III E/Centaur increases the maximum payload to almost 700 kg for a flight time of 6.5 years and the 30 day orbit period. The Shuttle/Centaur/HE BII can deliver no more than 650 kg into this orbit using a five-year transfer. The Shuttle/Centaur/SEP (20 kw) combination puts a 750 kg spacecraft into a 30 day orbit using a 4.2 year flight time. Add about 75 days more for earth-storable propellants.

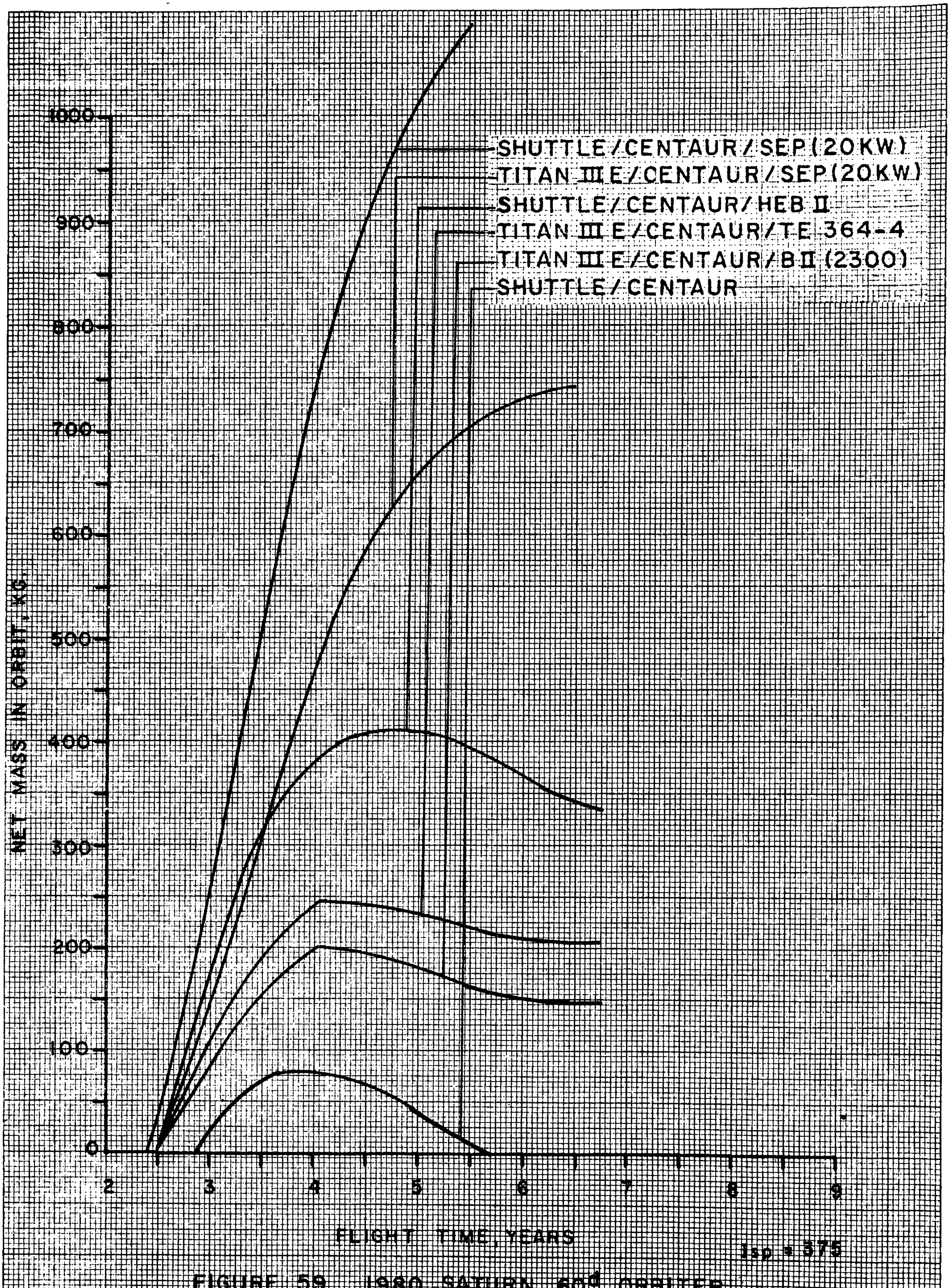


FIGURE 59. 1980 SATURN 60^{d} ORBITER

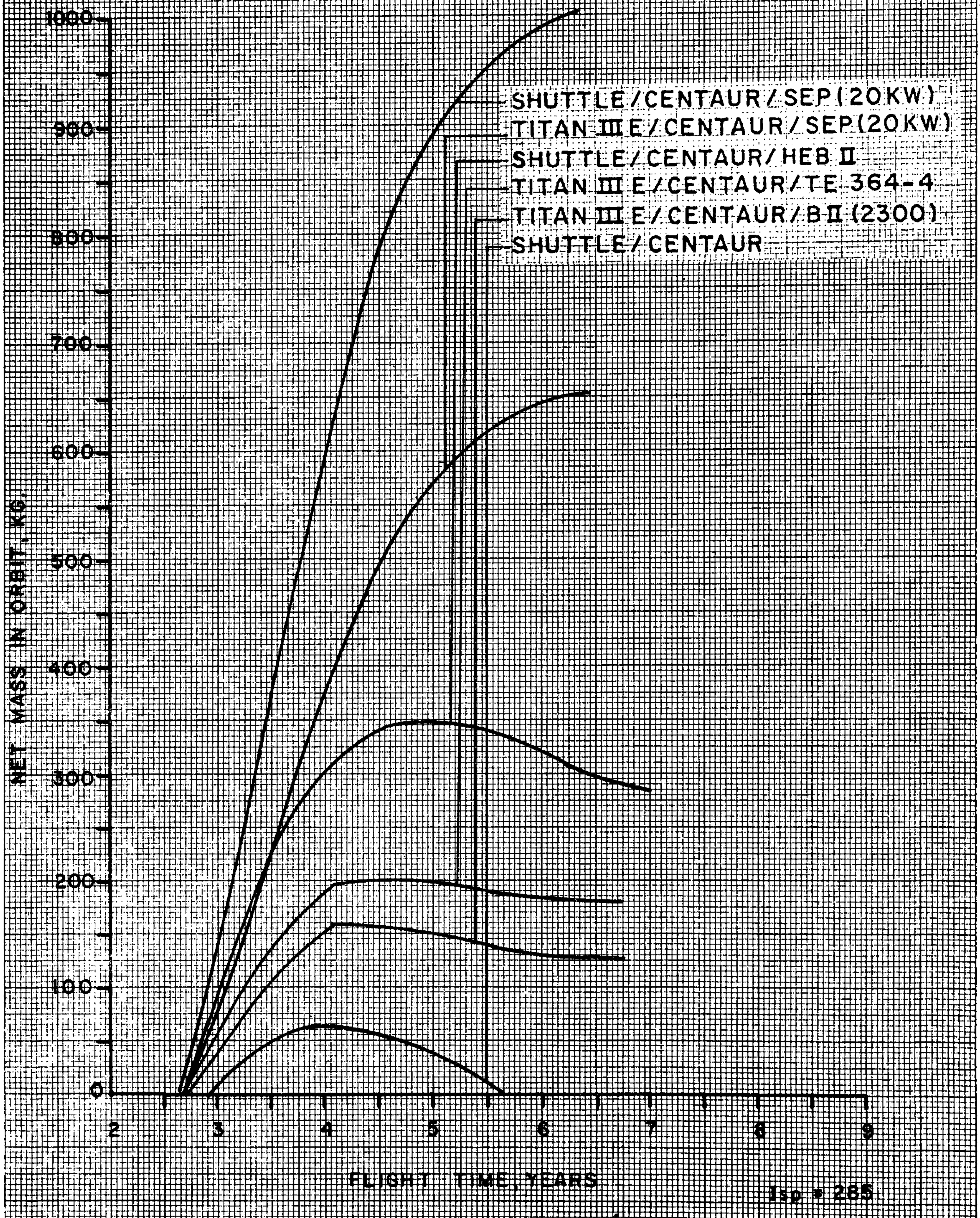


FIGURE 60. 1980 SATURN 60^{d} ORBITER

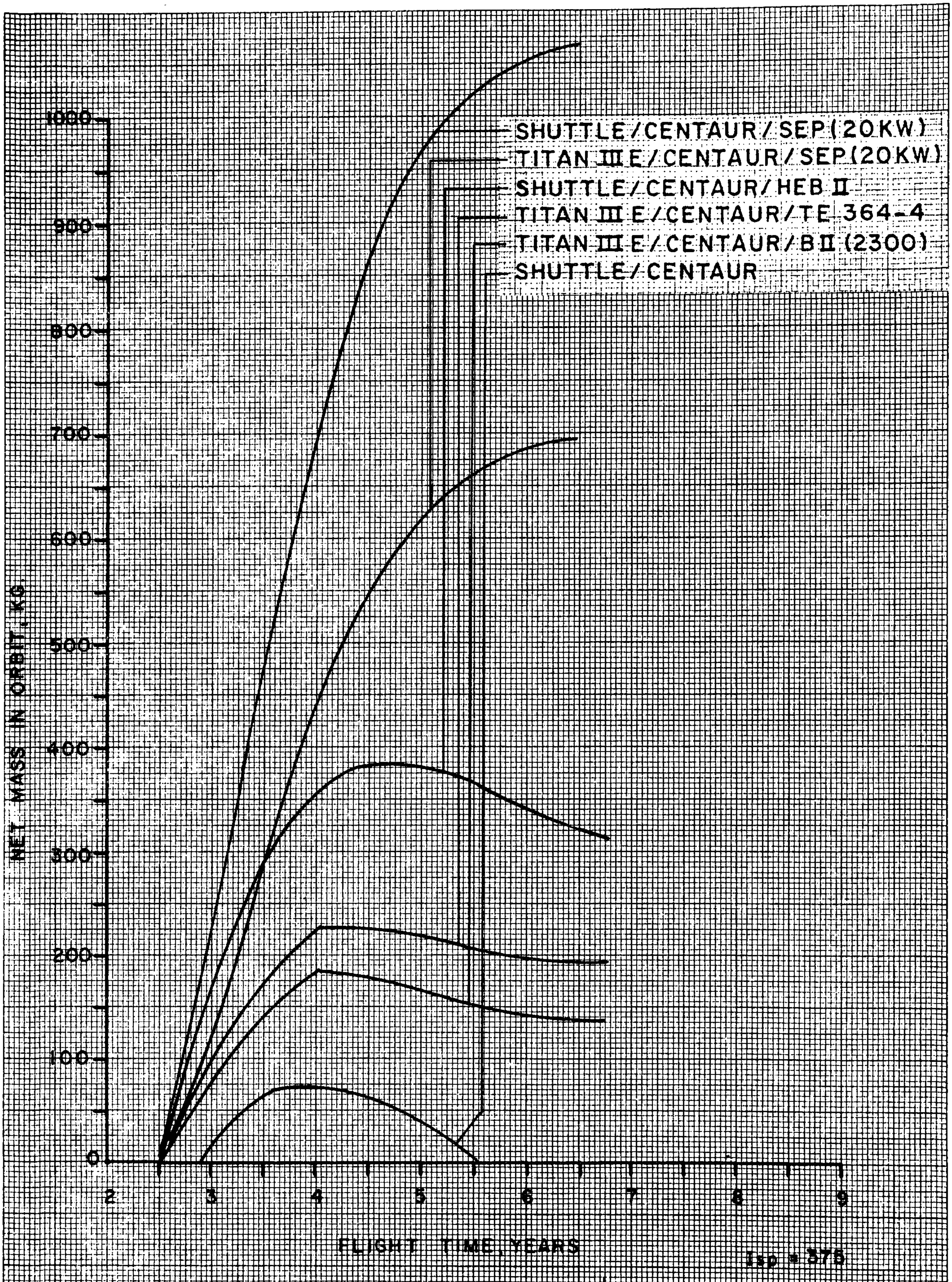


FIGURE 61. 1980 SATURN 30^d ORBITER

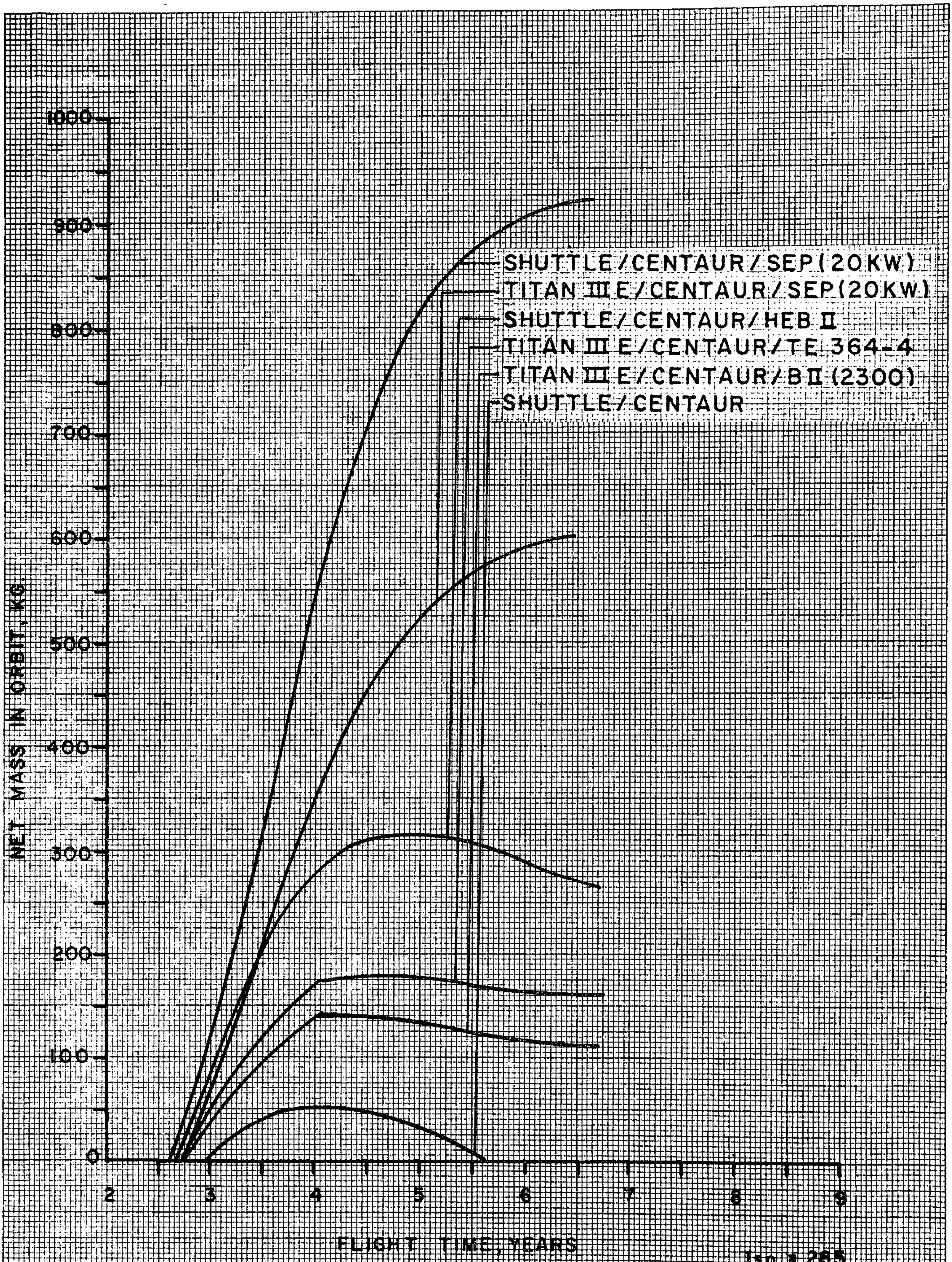


FIGURE 62. 1980 SATURN 30^{d} ORBITER

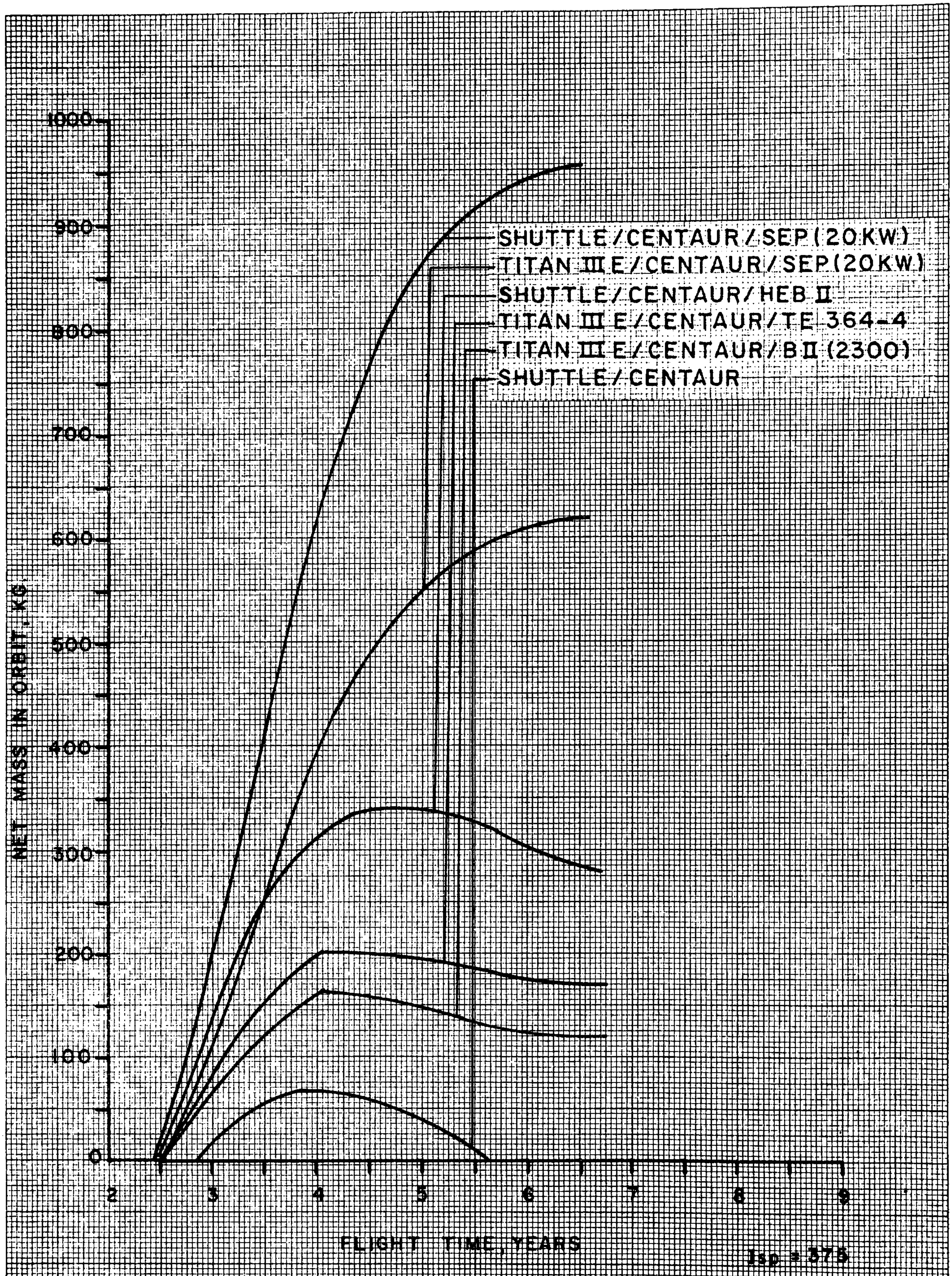


FIGURE 63. 1980 SATURN 15^d ORBITER

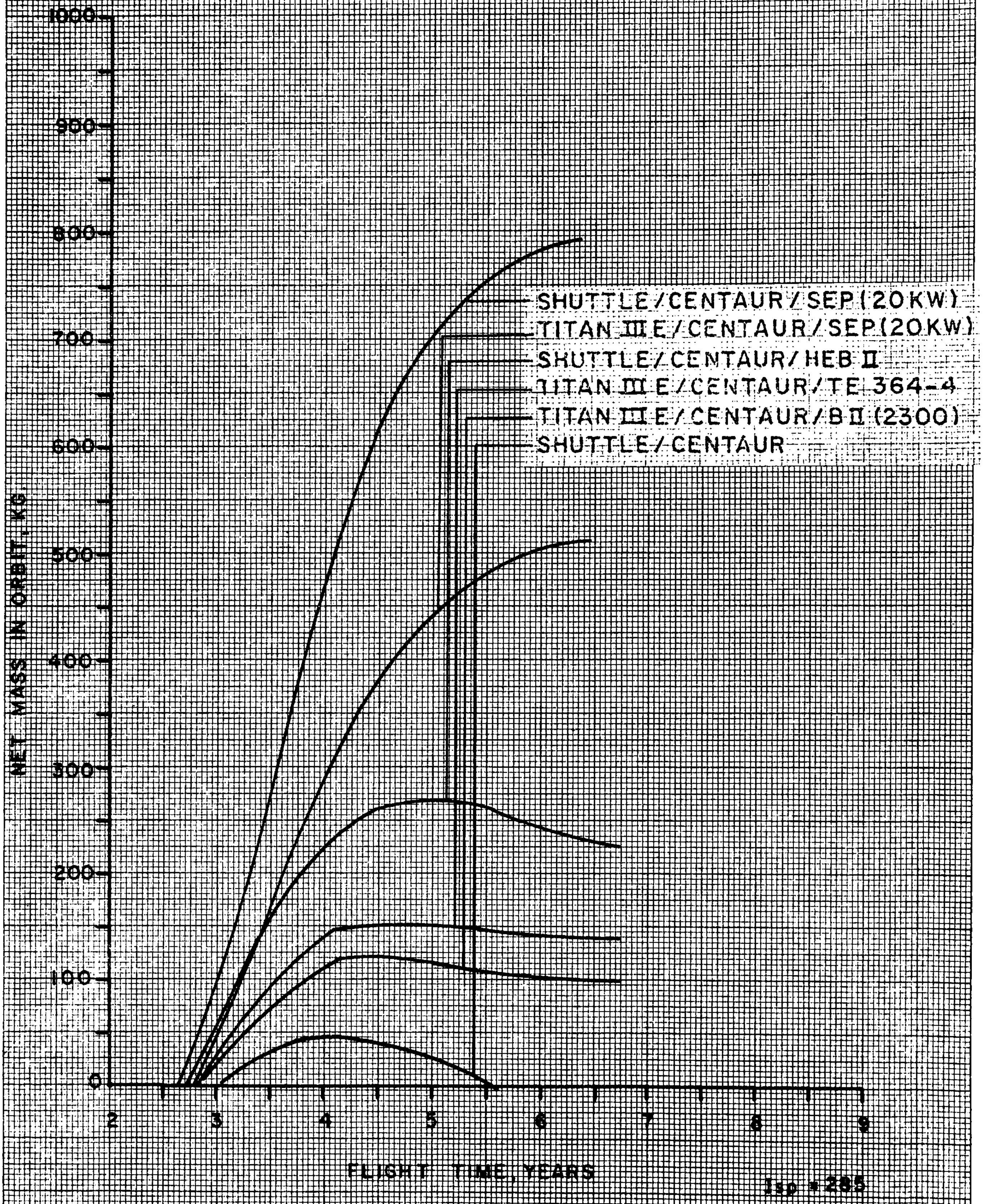


FIGURE 64. 1980 SATURN 15d ORBITER

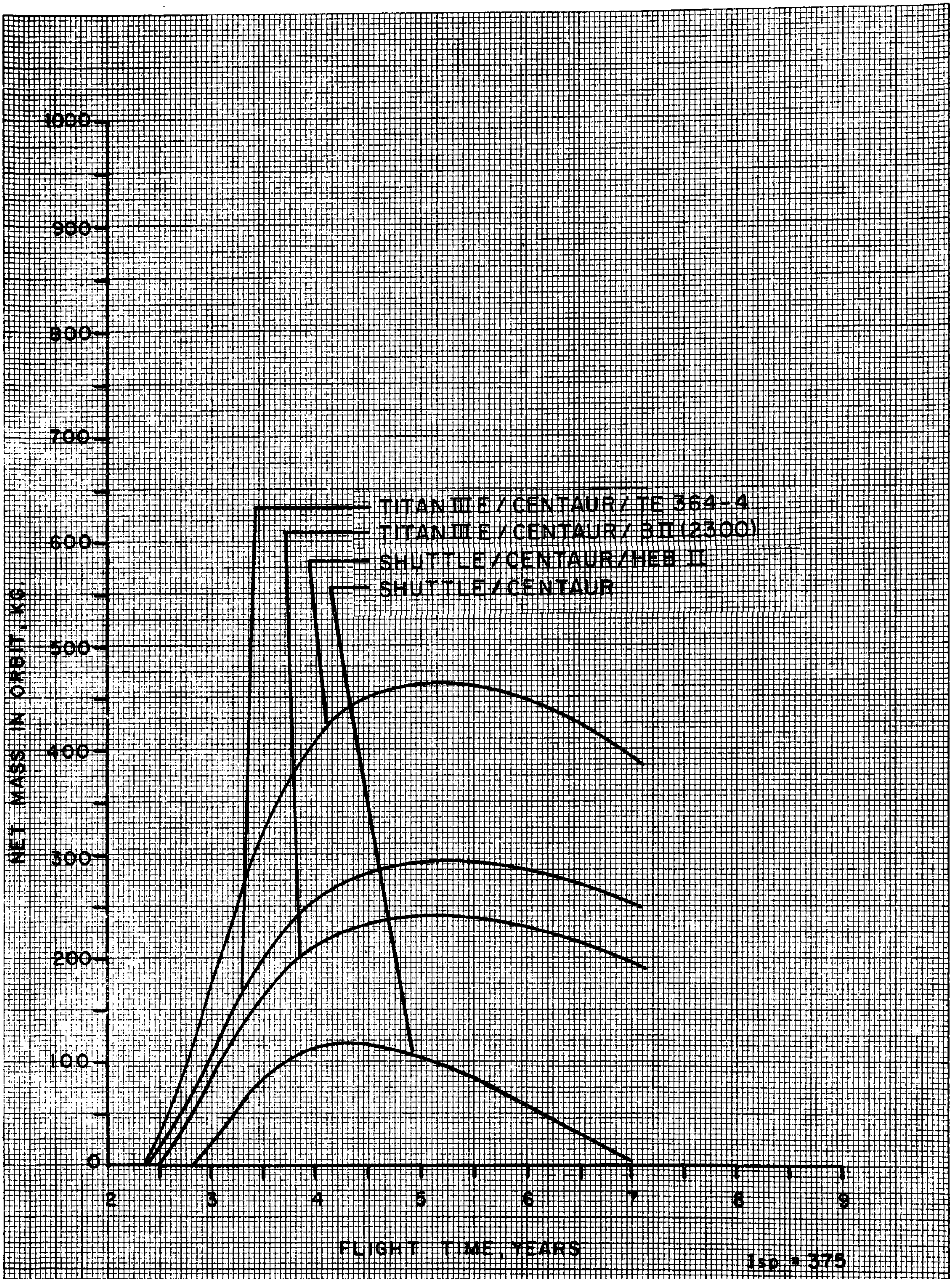


FIGURE 65. 1981 SATURN 60^d ORBITER

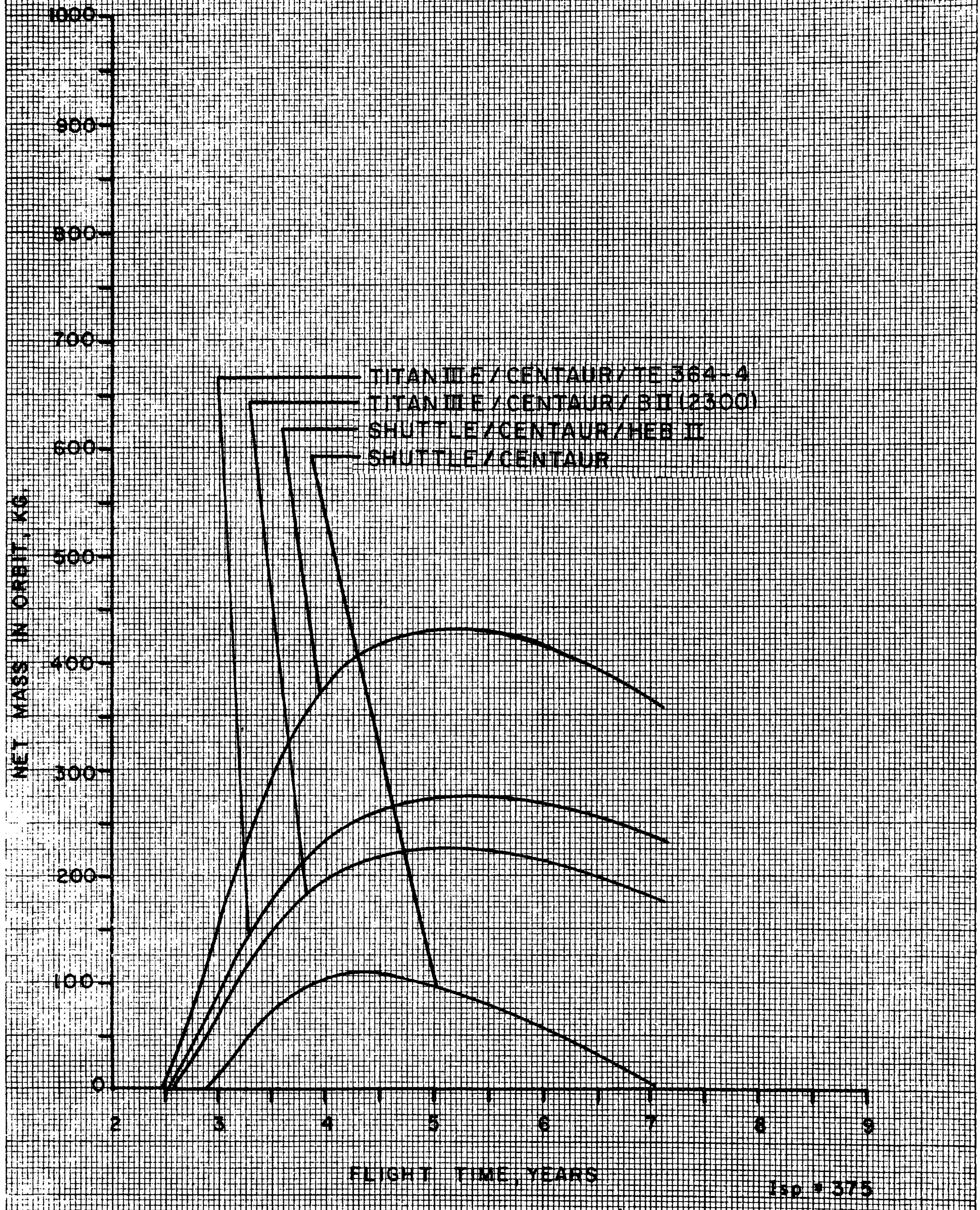


FIGURE 66. 1981 SATURN 30^d ORBITER

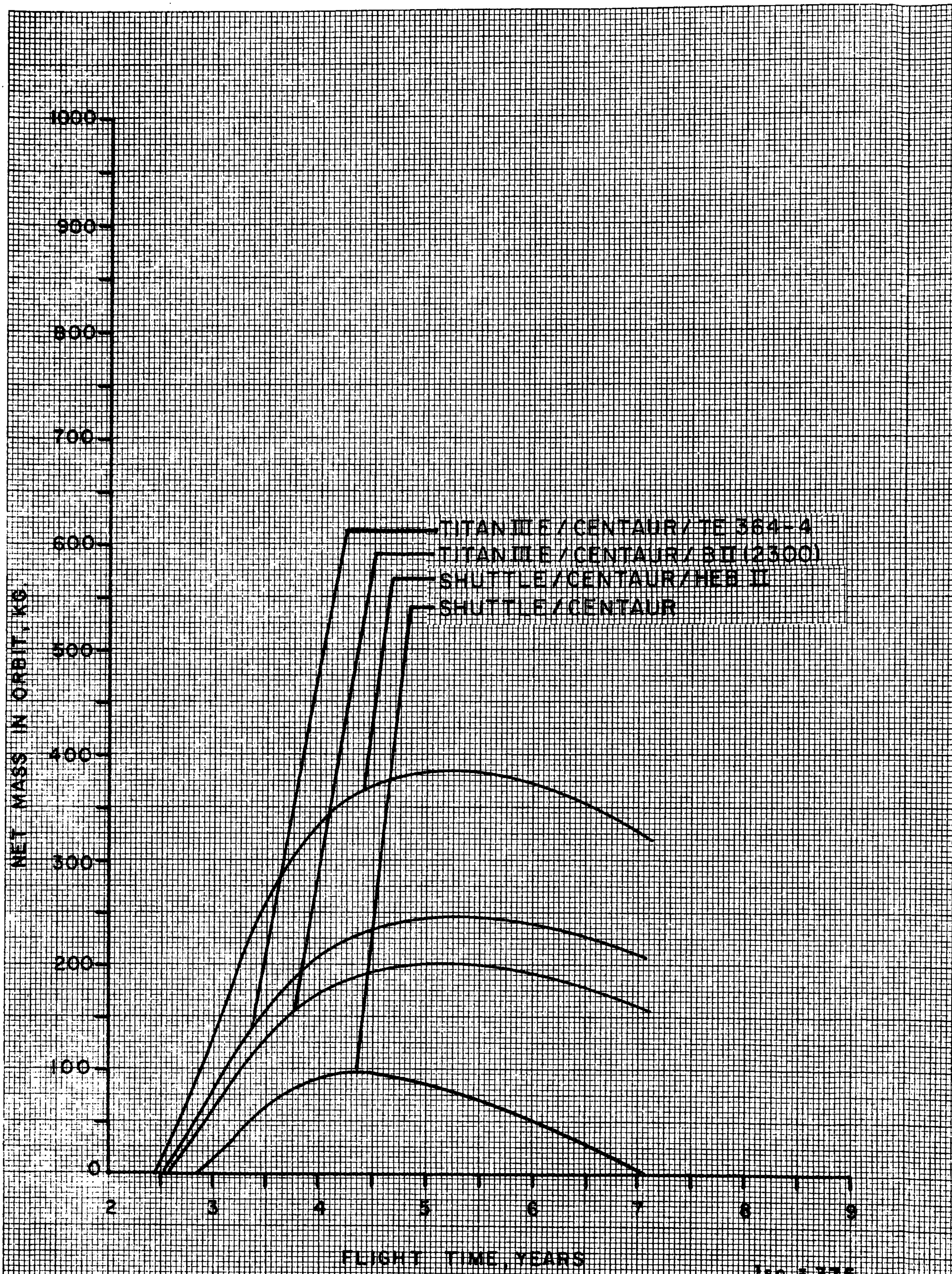


FIGURE 67. 1981 SATURN 15° ORBITER

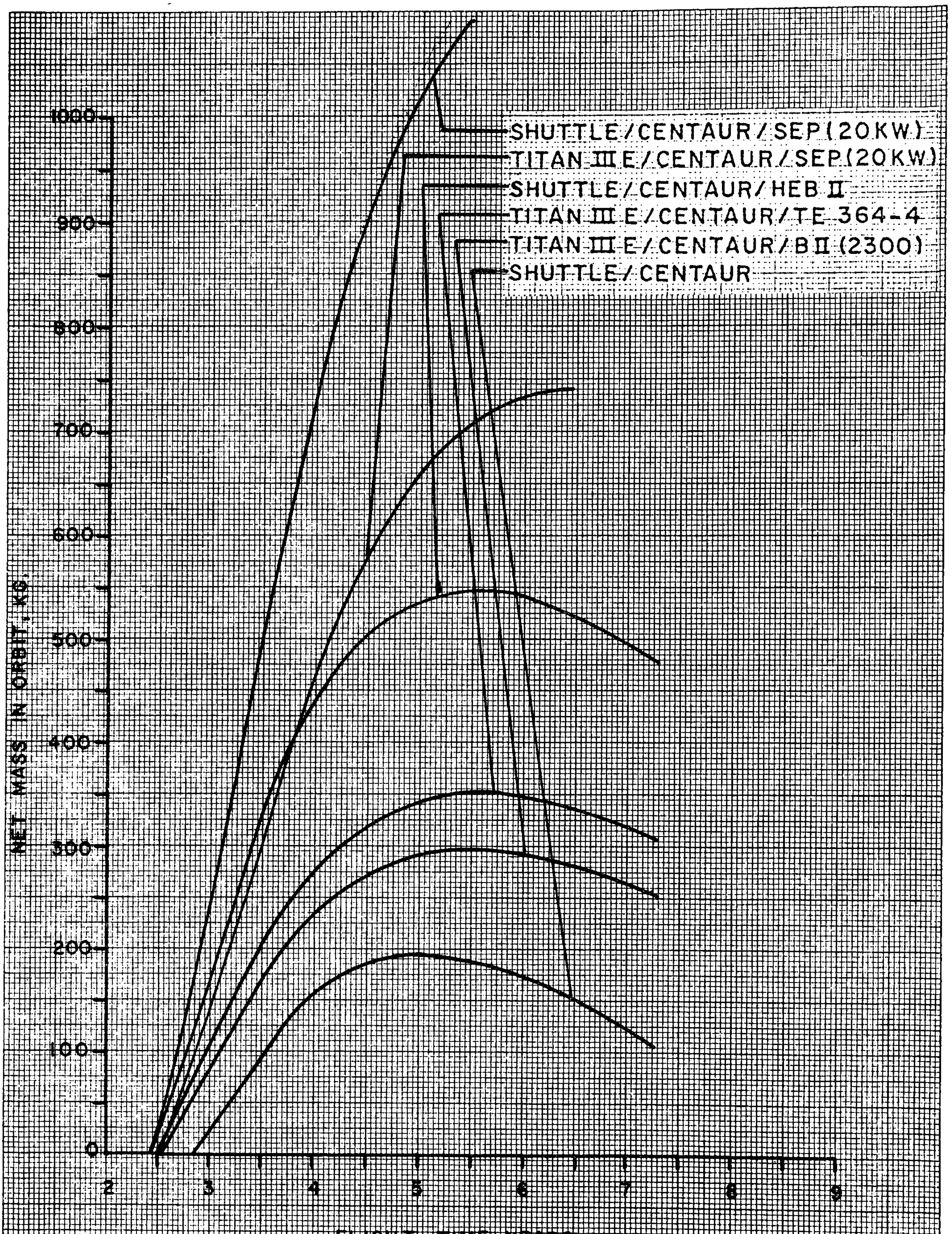


FIGURE 68. 1982 SATURN 60^d ORBITER

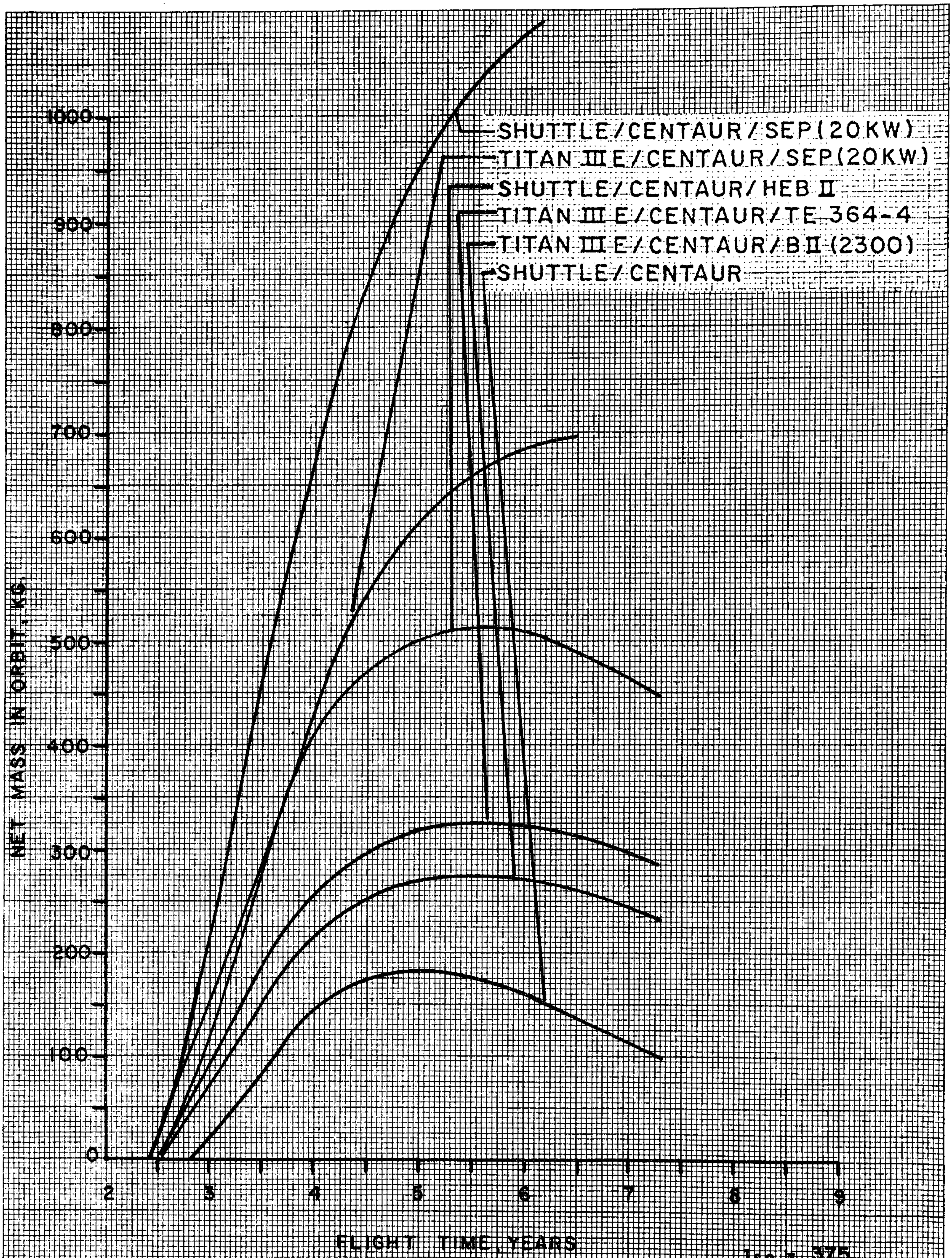


FIGURE 69. 1982 SATURN 30^d ORBITER

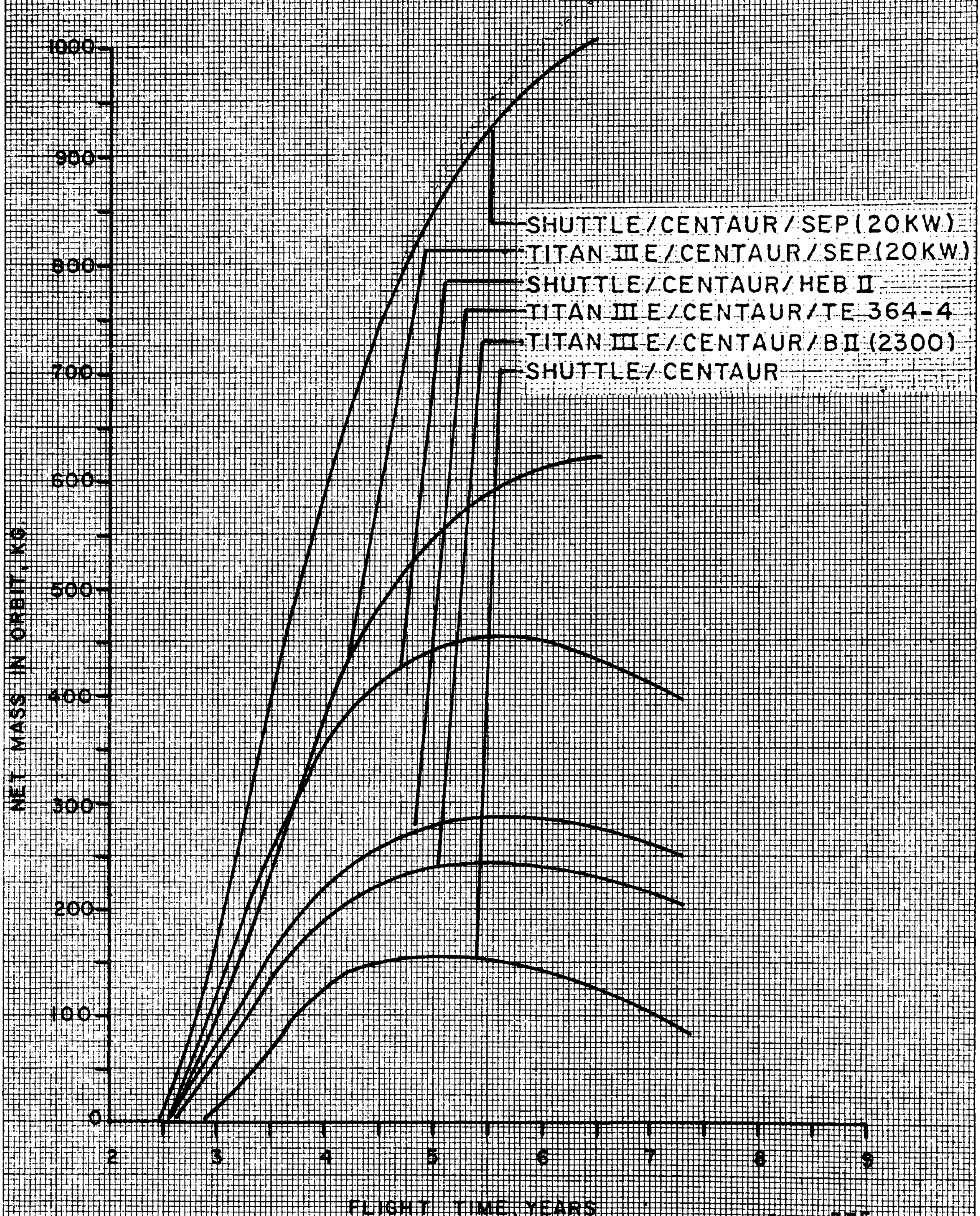


FIGURE 70. 1982 SATURN 15d ORBITER

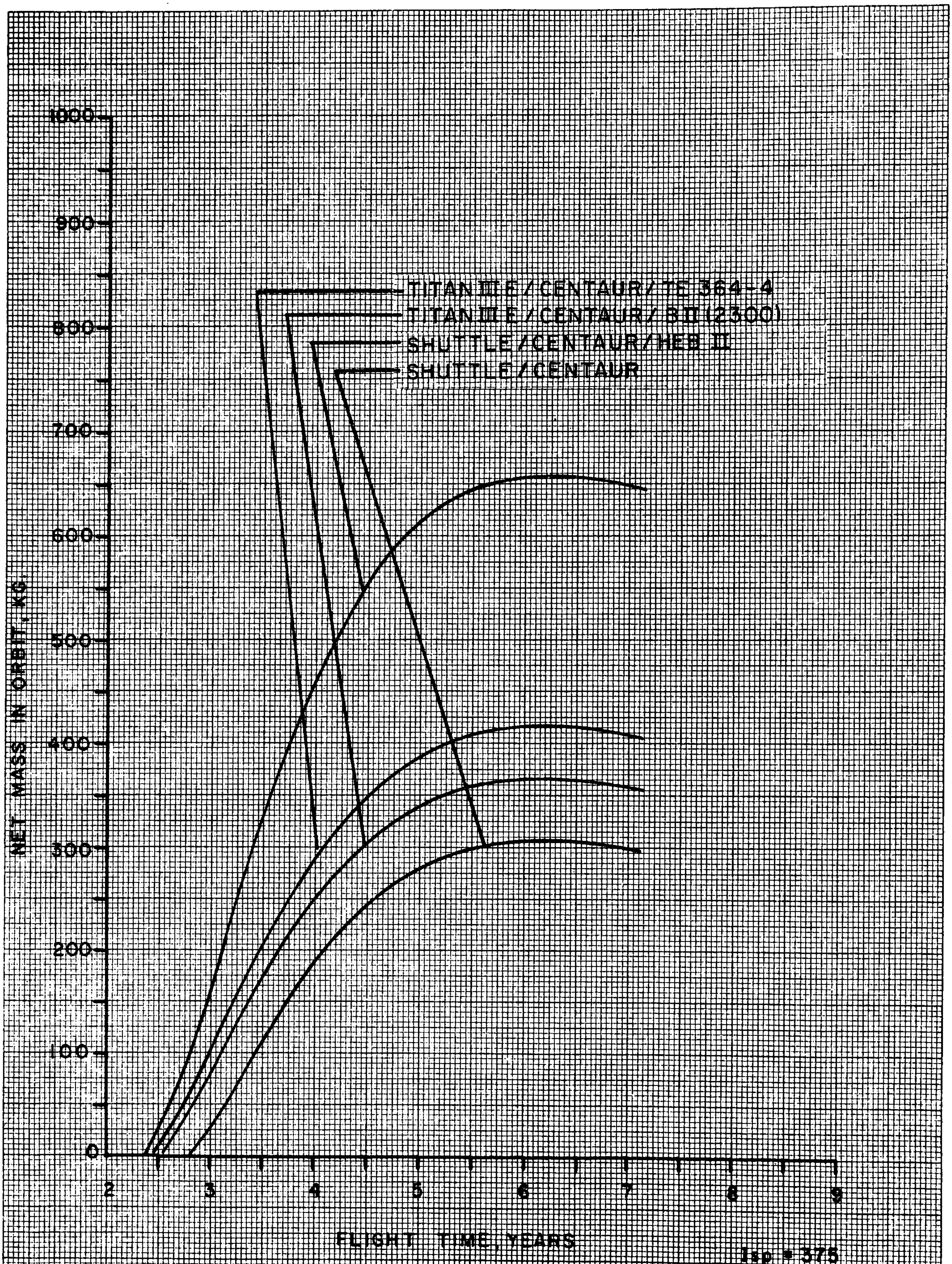


FIGURE 71. 1983 SATURN 60^d ORBITER

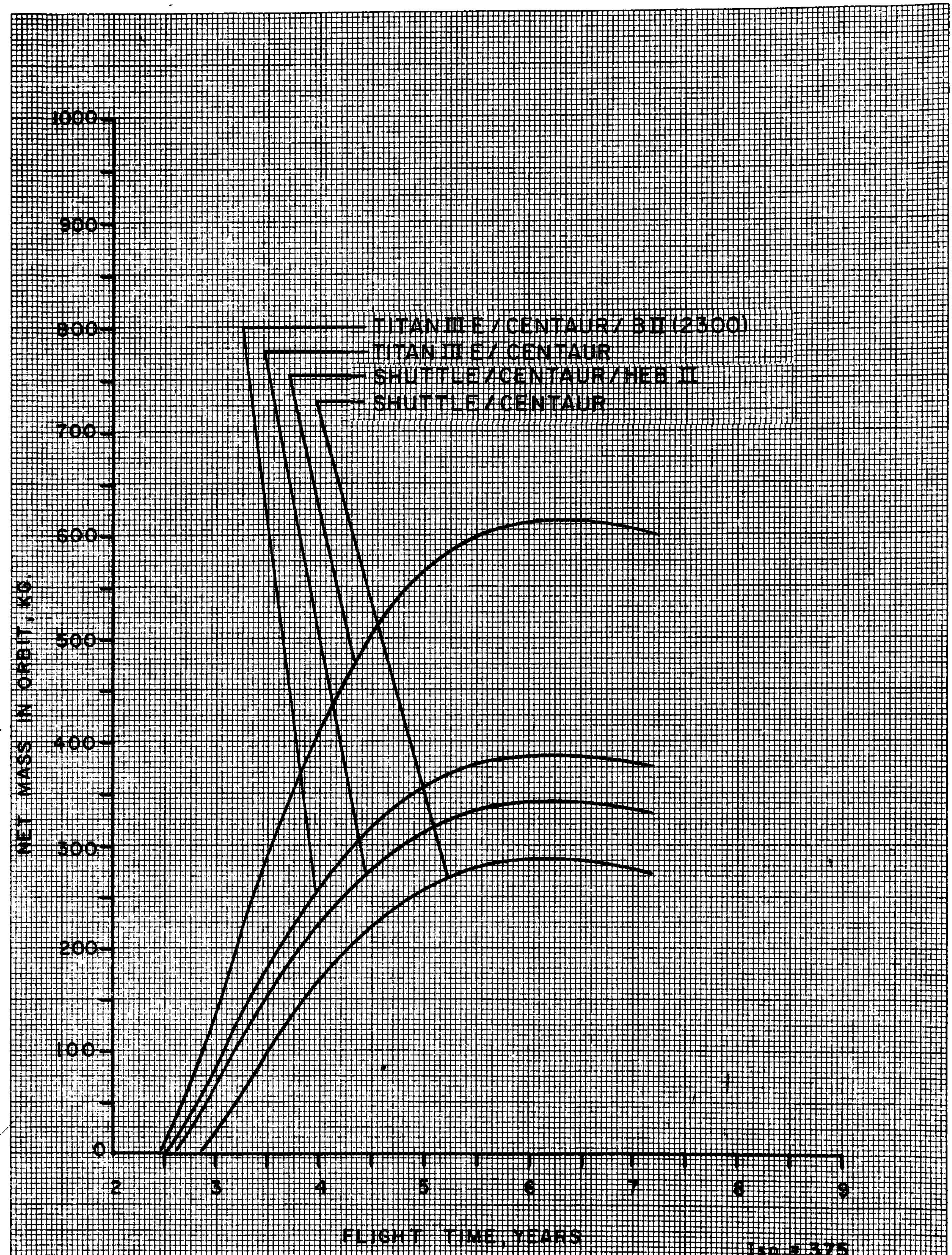


FIGURE 72. 1983 SATURN 30^d ORBITER

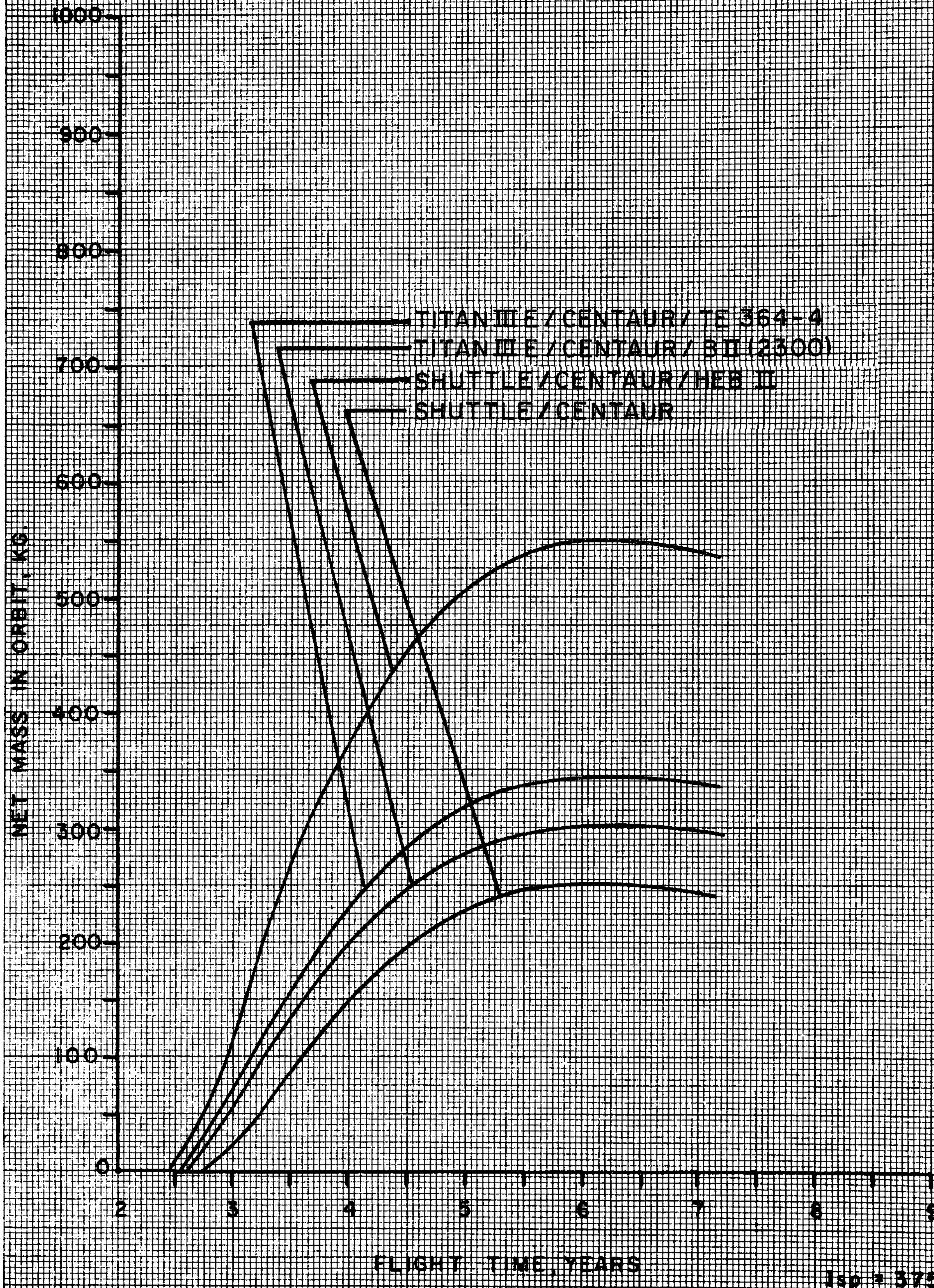


FIGURE 73. 1983 SATURN 15^d ORBITER

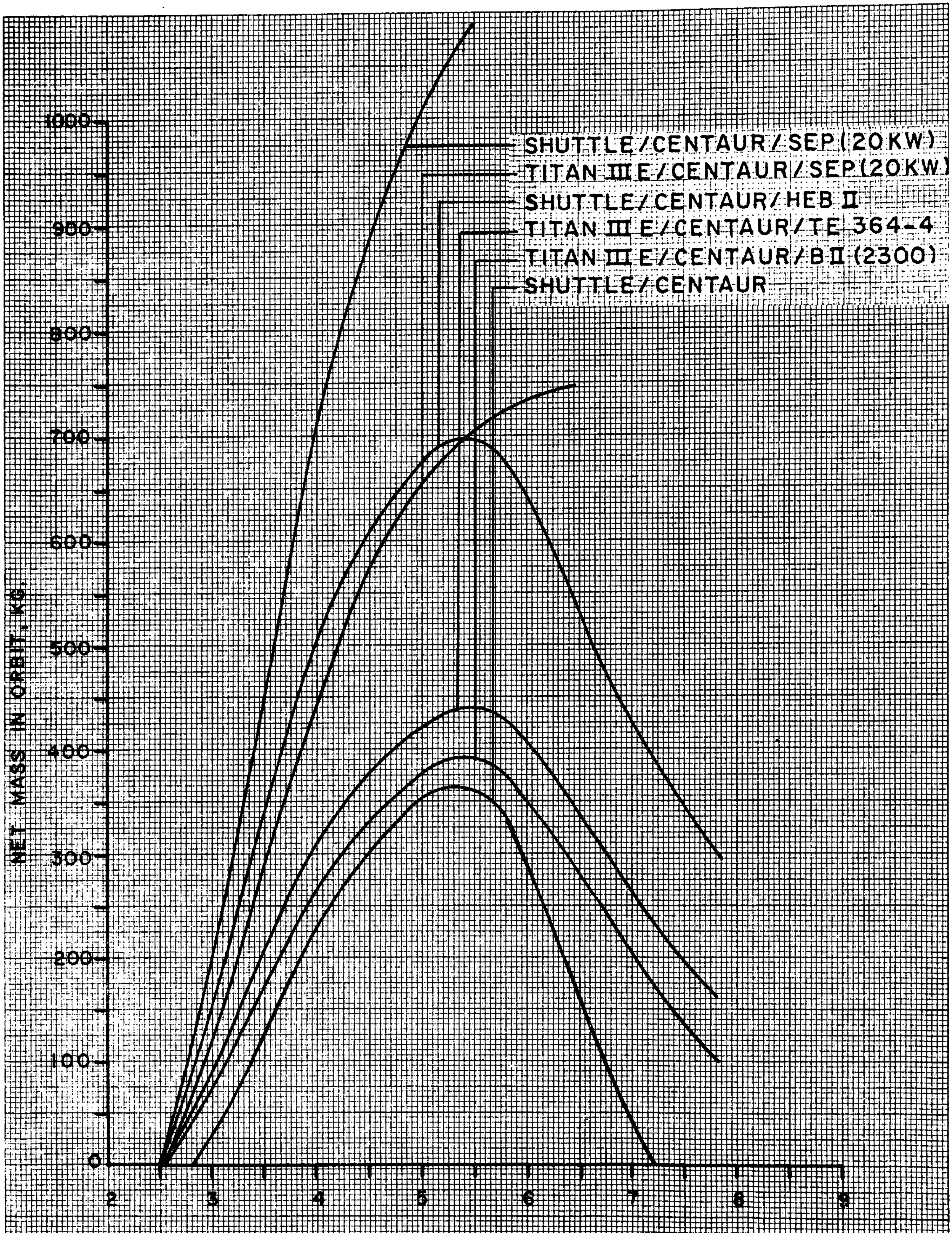


FIGURE 74. 1985 SATURN 60° ORBITER

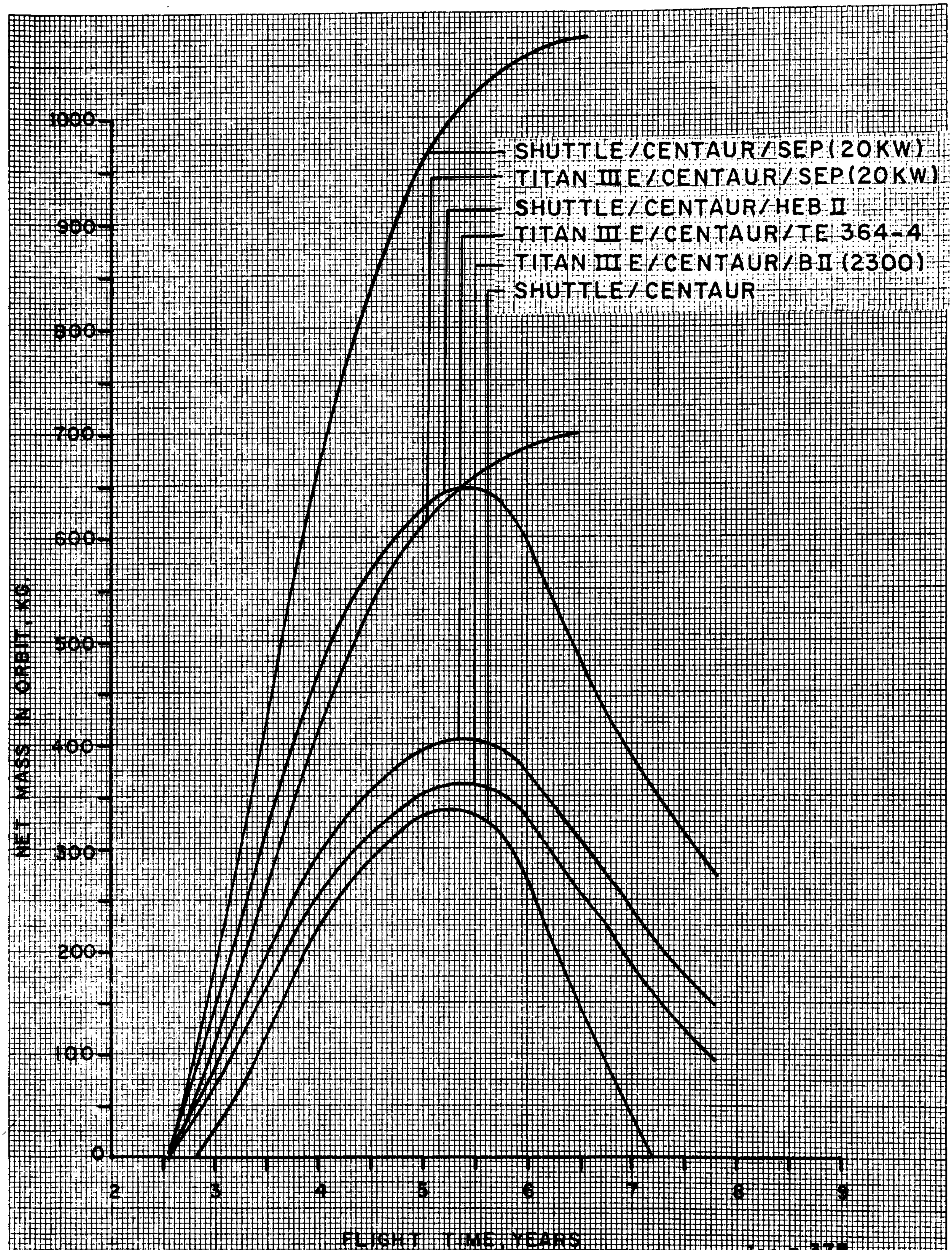


FIGURE 75. 1985 SATURN 30^d ORBITER

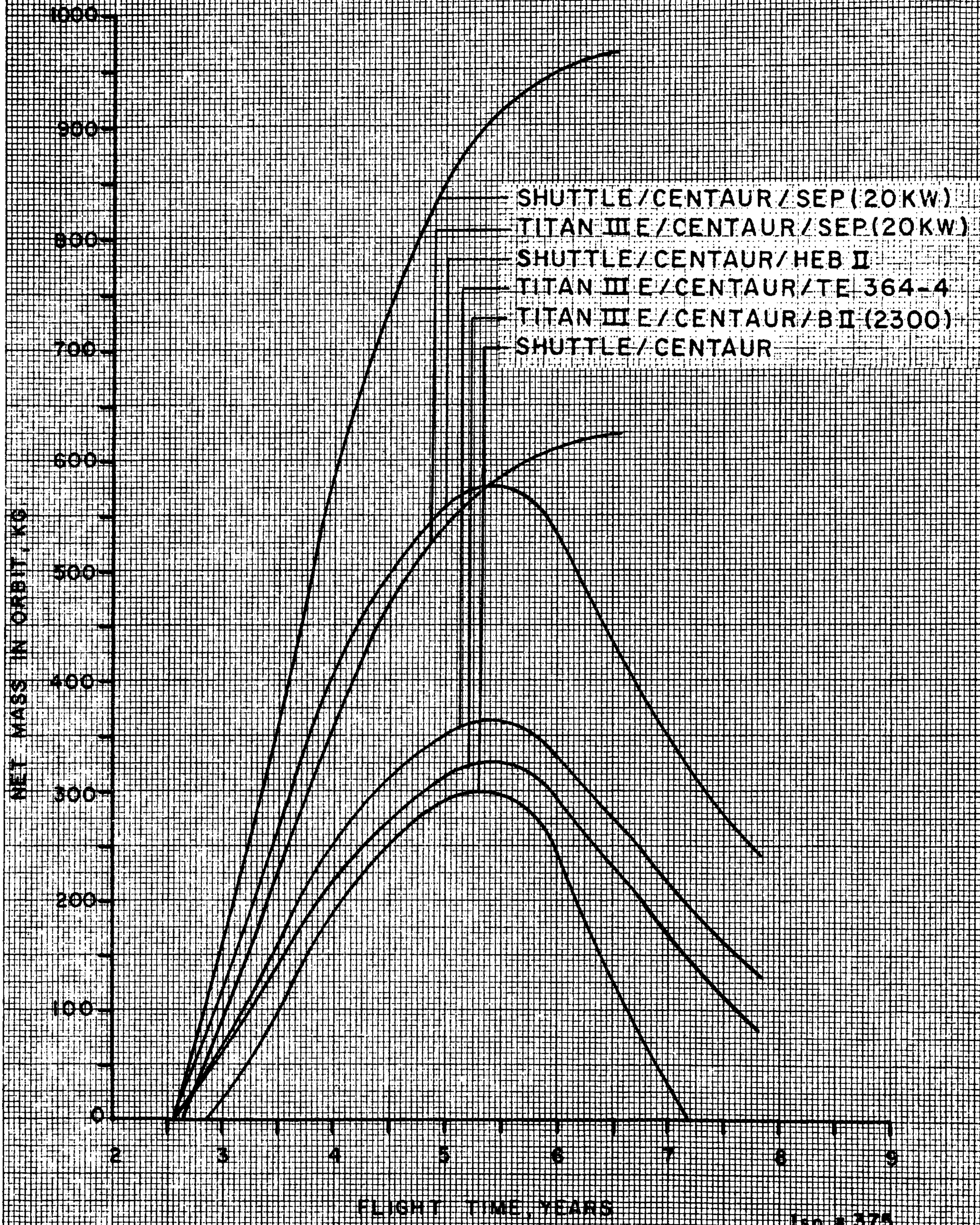


FIGURE 76. 1985 SATURN 15^d ORBITER

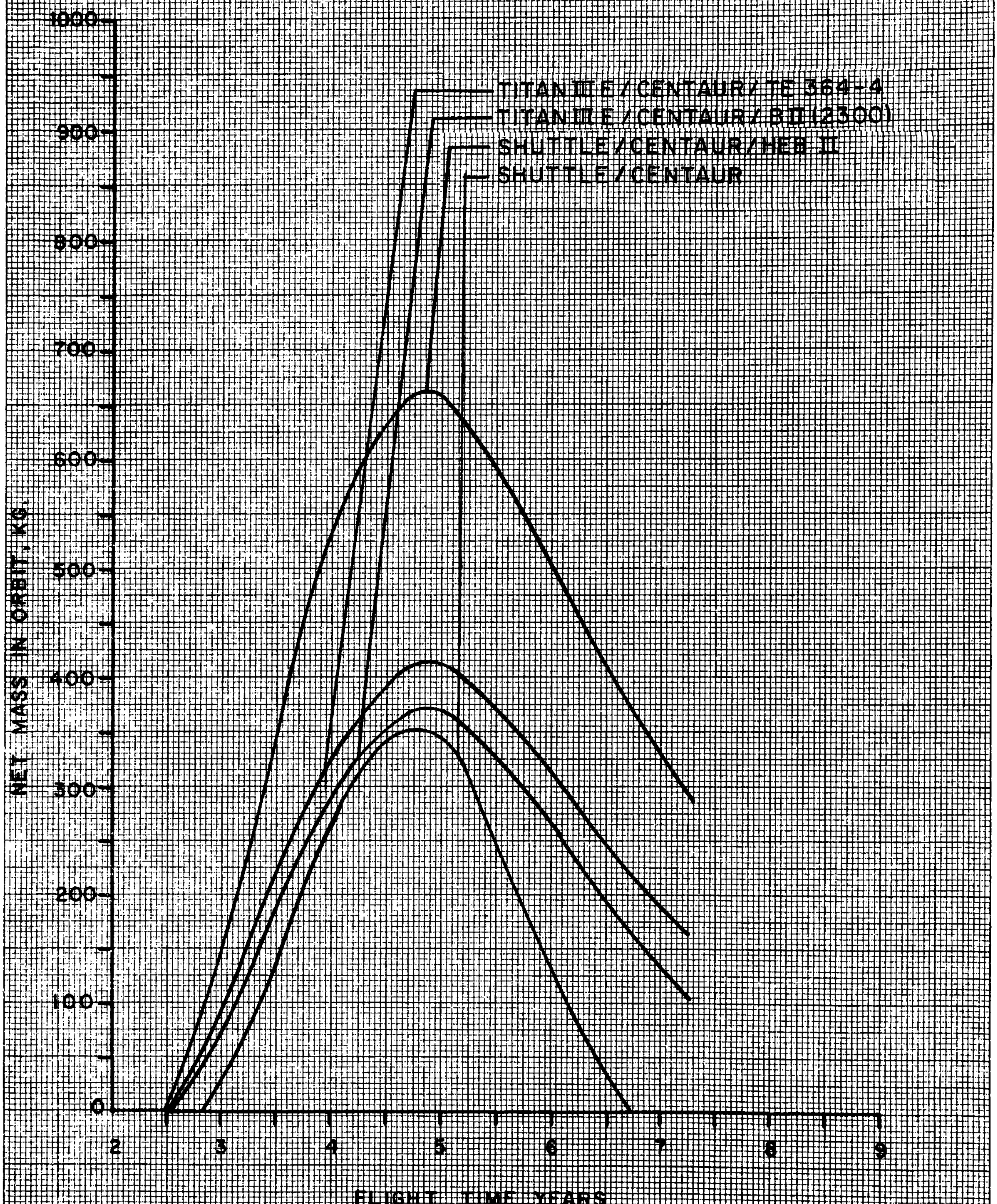


FIGURE 77. 1986 SATURN 60^d ORBITER

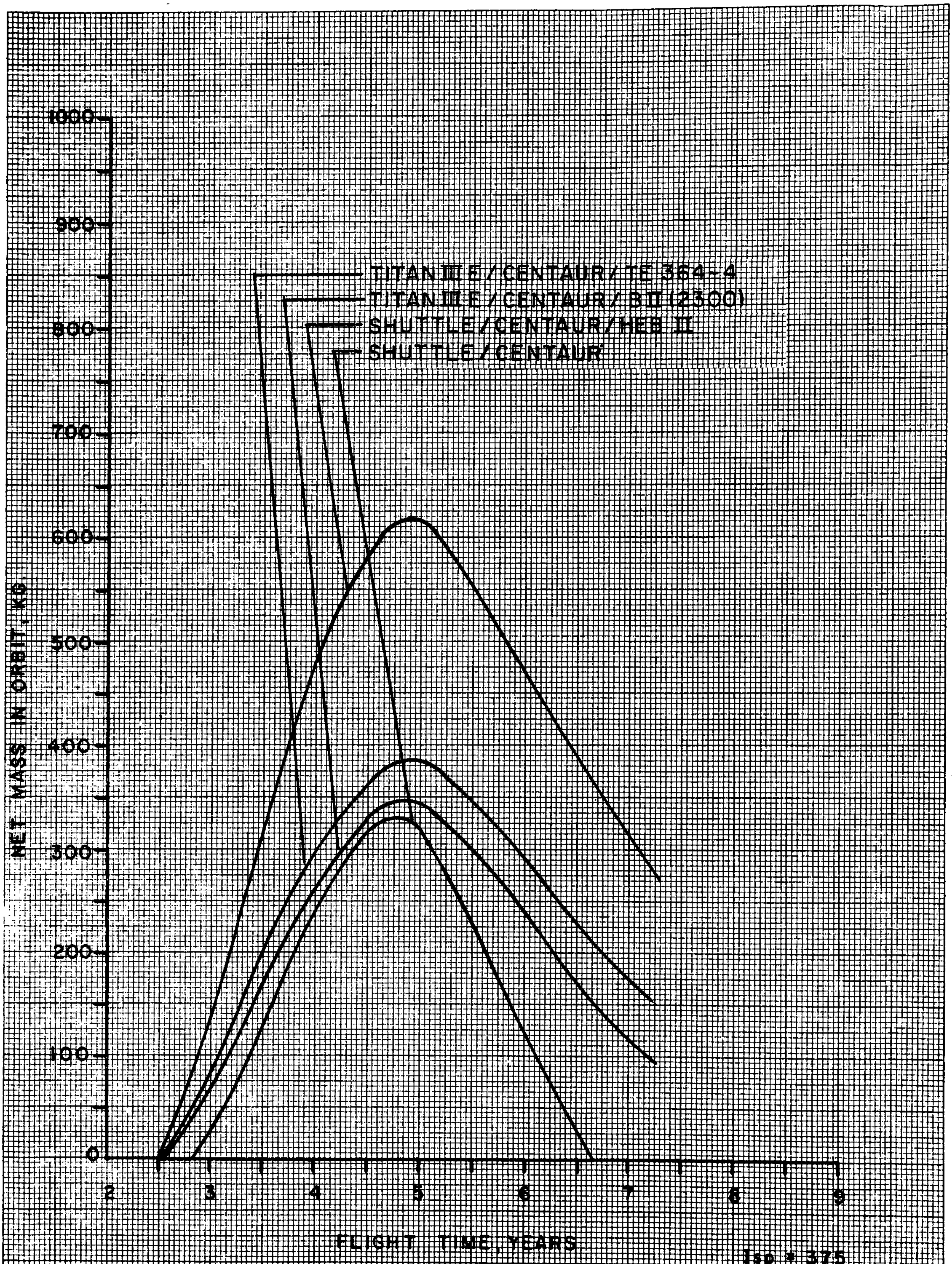


FIGURE 78. 1986 SATURN 30[d] ORBITER

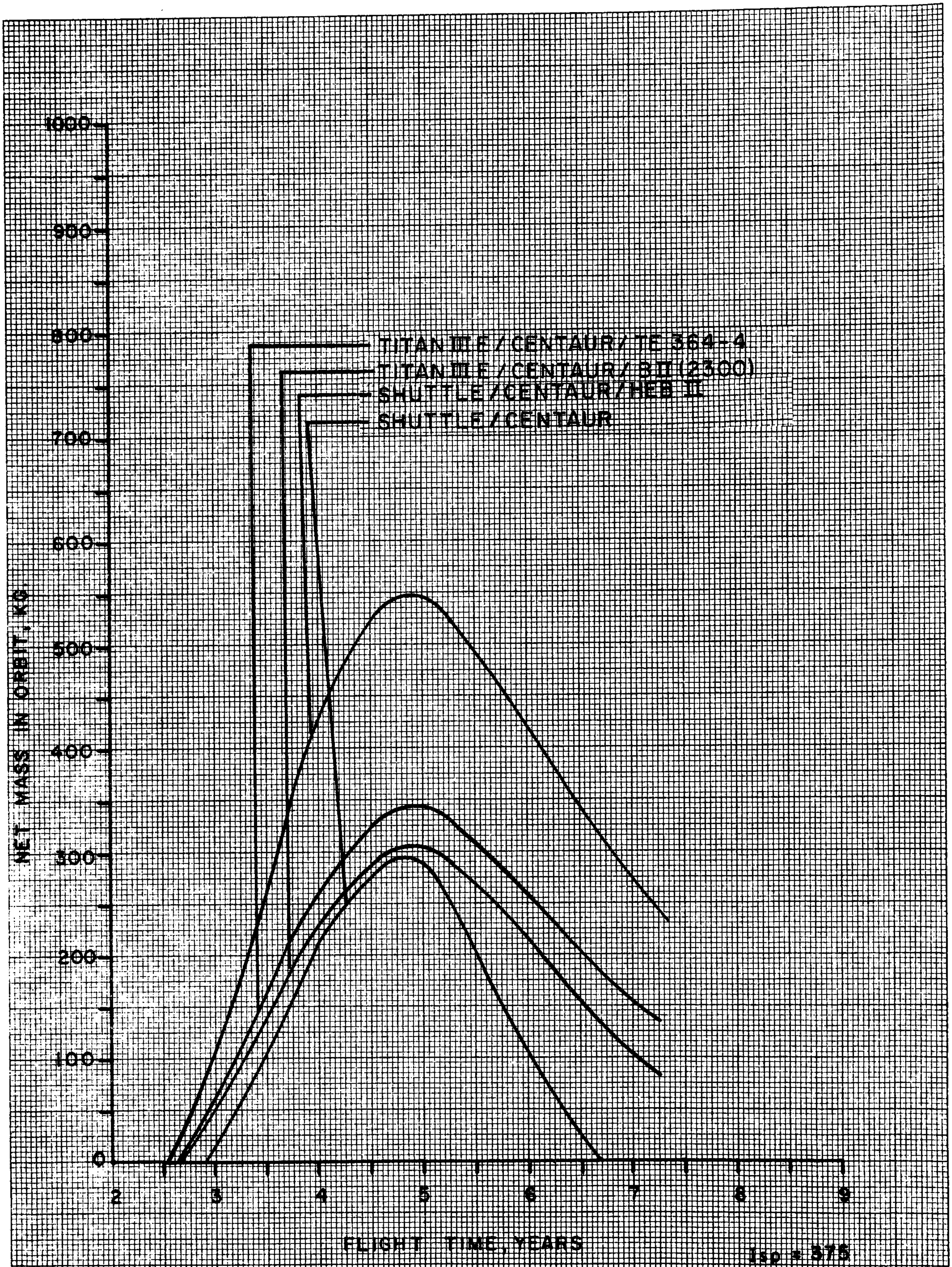


FIGURE 79. 1986 SATURN 15° ORBITER

3.5 Uranus Missions: 1985

Because it takes Uranus 84 years to go around the sun, the one opportunity studied (1985) should be typical of all launches in the 1980's. Direct ballistic flyby missions to Uranus in 1985 can be more easily accomplished by using the Shuttle/Centaur. A 750 kg spacecraft requires a flight time of about 6.2 years if a 20 kw SEP stage is used or 7.9 years with a HEB II upper stage. At similar flight times but using a Titan III E/Centaur as the launch vehicle, the flyby payload is only 450 kg with either SEP or the chemical upper stage. Even the best orbiter case, the 60 day period orbit with a periapse of 1.2 Uranus radii using a Shuttle launched SEP stage, requires 7 years for 300 kg net mass in orbit and more than ten years for 750 kg.

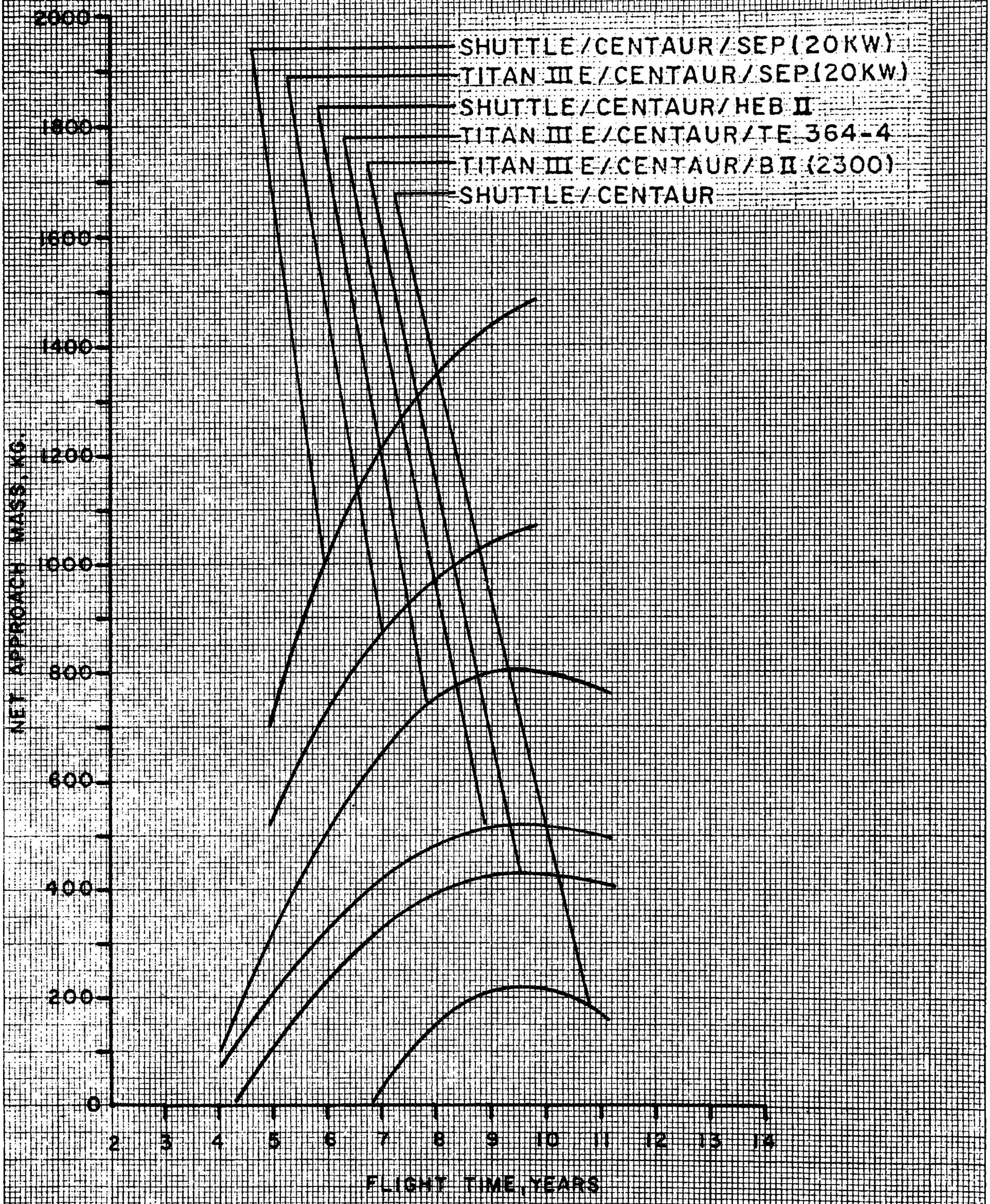


FIGURE 80. 1985 URANUS FLYBY

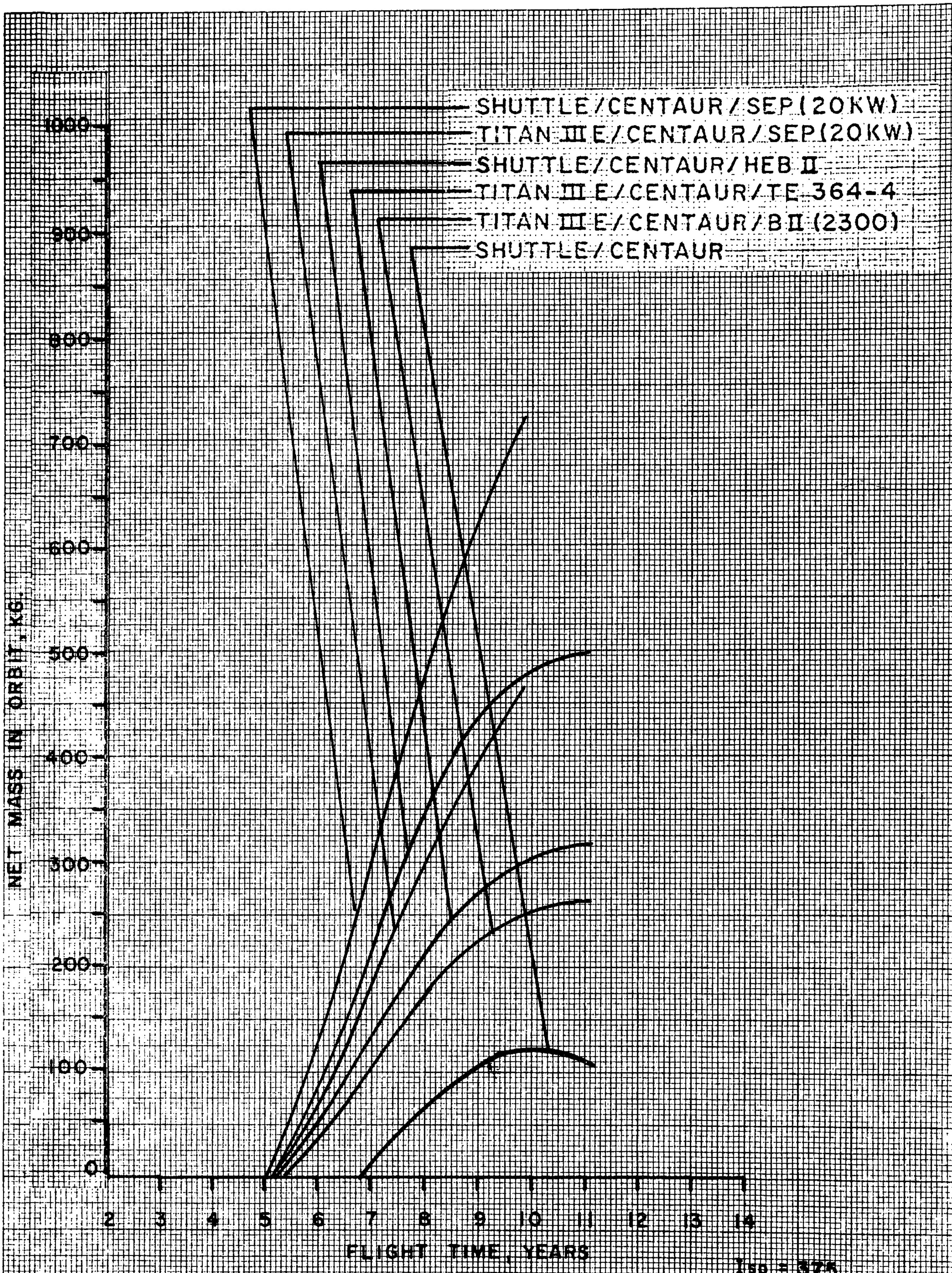


FIGURE 81. 1985 URANUS 60° ORBITER

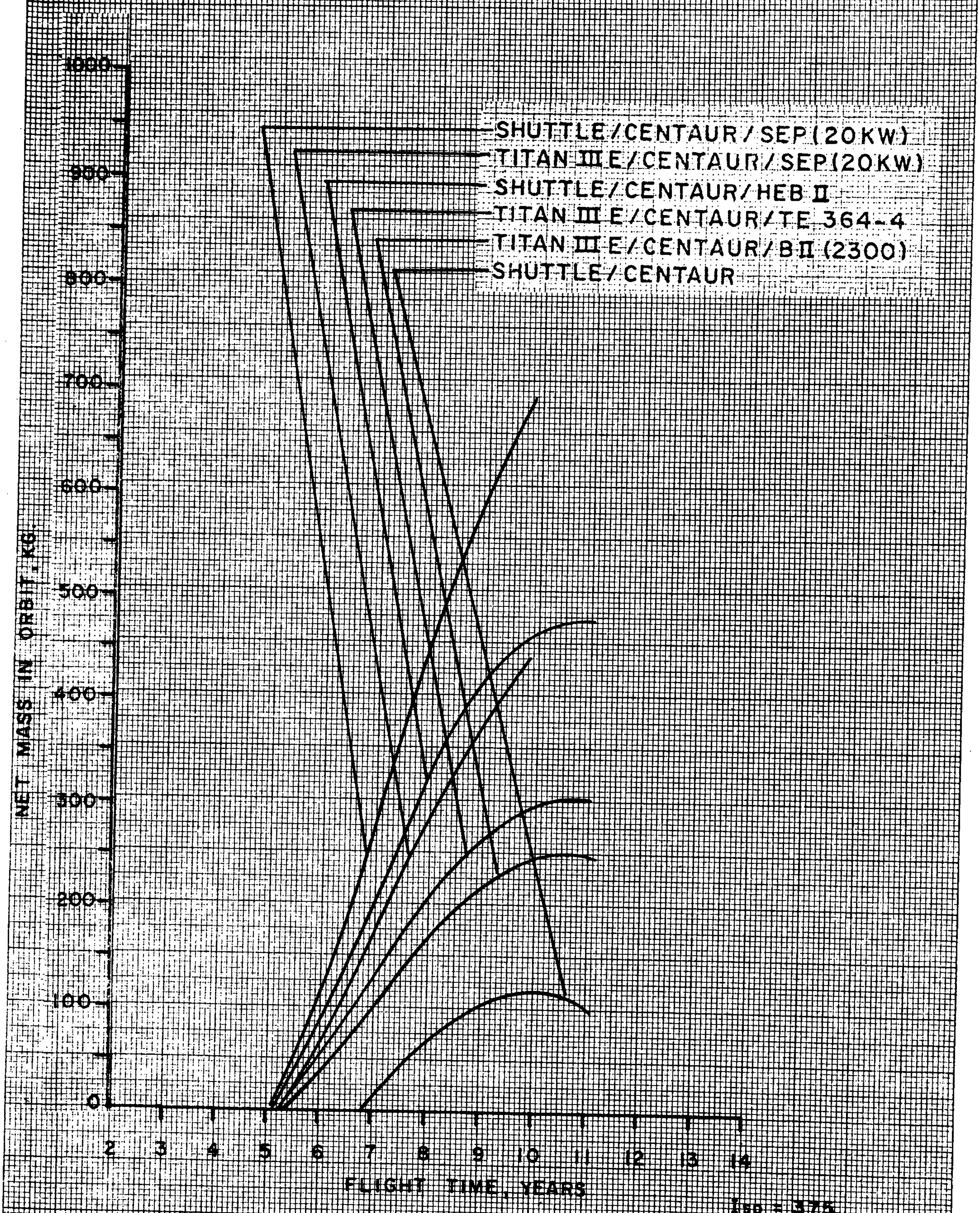


FIGURE 82. 1985 URANUS 15° ORBITER

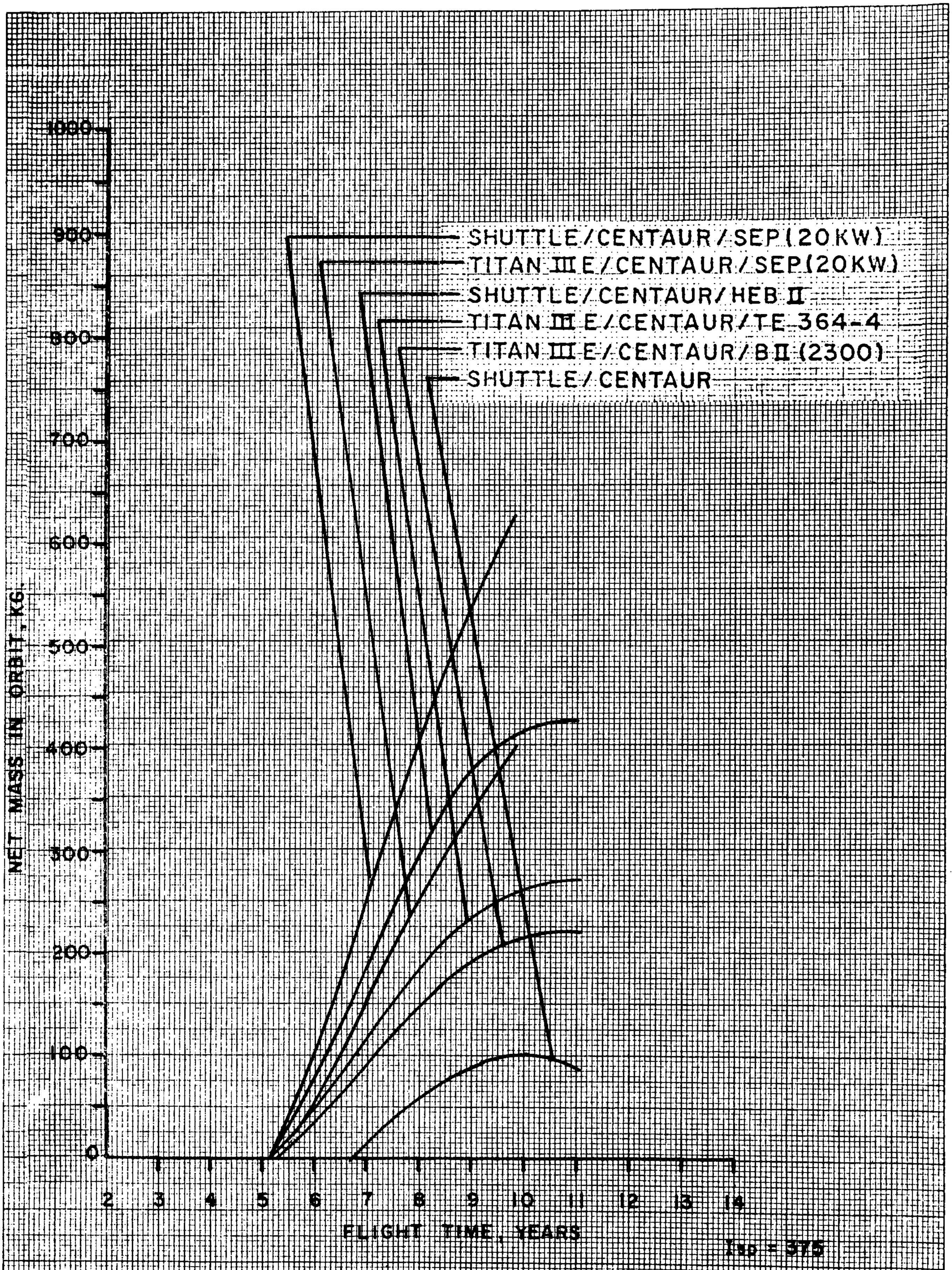


FIGURE 83. 1985 URANUS 5d ORBITER

REFERENCES

Chadwick, J.W. (1968), "Jettison Weight Function for Liquid Propellant Stages", Battelle Memorial Institute, BMI-NLVP-68-116.

Hahn, D.W. and F.T. Johnson (1971), "Chebychev Trajectory Optimization Program (Chebytop II)", Boeing Company, D-180-12916-1.

Horsewood, J.L. and F.I. Mann (1970), "Optimum Solar Electric Interplanetary Trajectory and Performance Data", NASA CR-1524.

NASA/OSSA (1972), "Launch Vehicle Estimating Factors", NASA-7100.5A.

NASA/OART (1969), "Planetary Flight Handbook", NASA SP-35, part 7.

Roth, R. et. al. (1968), "Space Research Conic Program, Phase III", Jet Propulsion Laboratory, 900-130.